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ReUsable Launch Vehicles (RLV)

AA/SS 4000 Series Briefing
August 3, 2001

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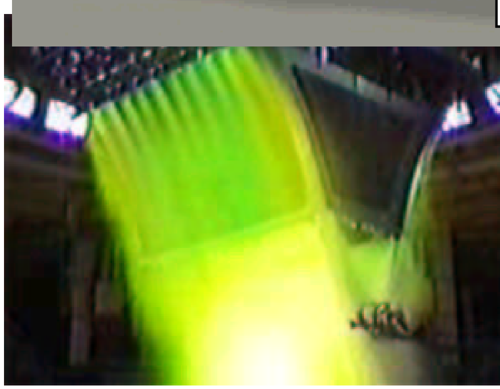
Space Systems Academic Group
Naval Postgraduate School



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Why Develop Re-Usable Launch Systems?

• **The surface of Earth lies at the bottom of a deep gravity well and a vast ocean of air**

...... the sheer speed required to attain orbit demands a very high order of launch vehicle performance.

• **Although US acquired capability to place payloads and people to orbit several decades ago**

.... space travel is still an enormously complex, expensive, and dangerous undertaking

• **Extremely high cost of space access presents tremendous limitation to large-scale space commercialization**

.... to achieve a profit, value of current commercial payloads must literally exceed their weights in gold

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Why Develop Re-Usable Launch Systems? (concluded)

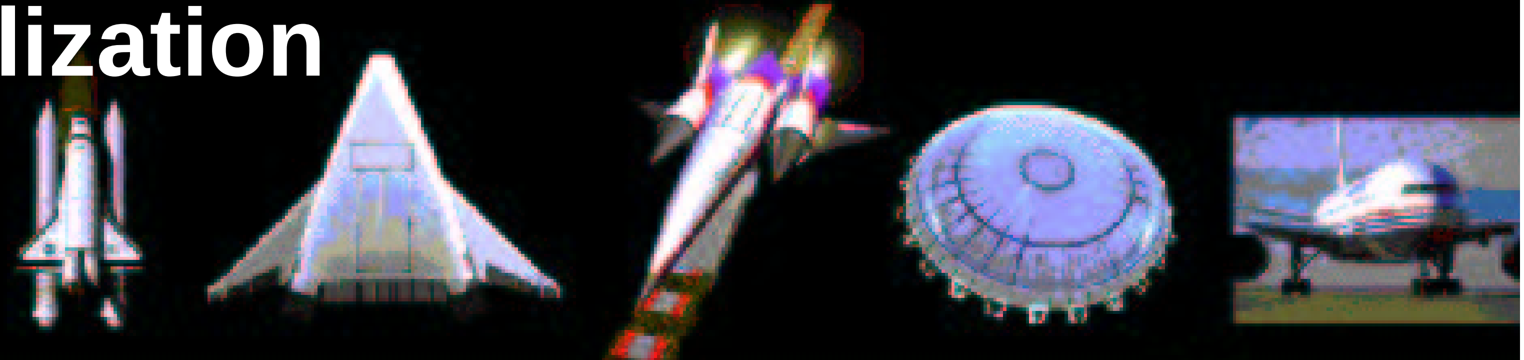
• **A NASA Study Conducted in 1992 concluded that in to achieve large-scale space commercialization and/or militarization, then we must**

-- 1) Reduce payload cost to low Earth orbit (LEO) from $20,000 /pound to $1000 /pound within 10-20 years

-- 2) to $100 /pound within 25-30 years

-- 3) and finallly, to tens of dollars /pound within 40-50 years.

Road Map For Large-Scale Space Industrialization



Timeframe	Today	1) 10 Years	2) 25 Years	3) 40 Years	Today
Launch Costs	$10,000/lb	$1,000/lb	$100/lb	$10/lb	$1/lb
Catastrophic Failure	1 in 200 Flights	1 in 10,000 Flights	1 in 1,000,000 Flights	1 in 1,000,000 Flights	1 in 2,000,000 Flights
Crew Escape	None	Yes	Yes	Not Required	Not Required
Fleet Flights Per Year	10	100	2,000	10,000	Millions
Turnaround Time	5 Months	1 Week	1 Day	2 Hours	1 Hour
People Required to Launch	170	10	2	None	None
Range Safety	Flight Unique	Mission Class Unique	Space Traffic Control	Aerospace Trafffic Control	Air Trafffic Contro

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NASA's Integrated Space Transportation Program (ISTP)

- **NASA's long-range investment strategy for safer, more reliable, and less expensive access to space**

  -- Enable U.S. aerospace industry to develop new, privately owned and operated space transportation NASA as a customer.

- **ISTP consists of 3 major programs:**

  -- Space Shuttle Safety Upgrades **(1st Generation)**

  -- Space Launch Initiative, Near-term business plan for NASA and its partners, Reusable Launch Vehicle (RLV) Program, **(2nd Generation)**

  -- Propulsion (ScramJet, combined-combustion cycle) Single Stage-to-Orbit (SSTO) Technologies, and In-Space Transportation Systems **(3rd Generation)**

Integrated Space Transportation Program

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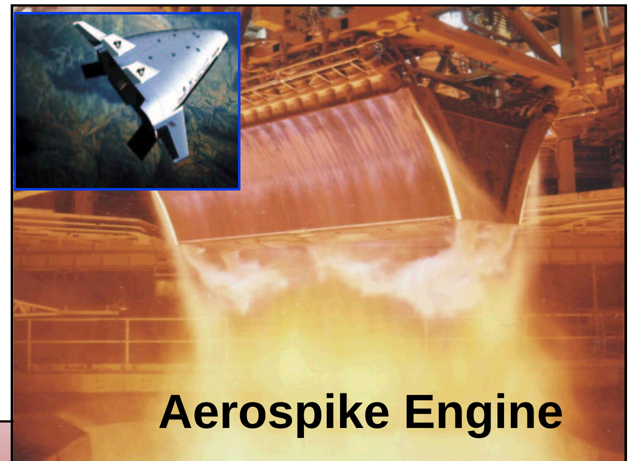
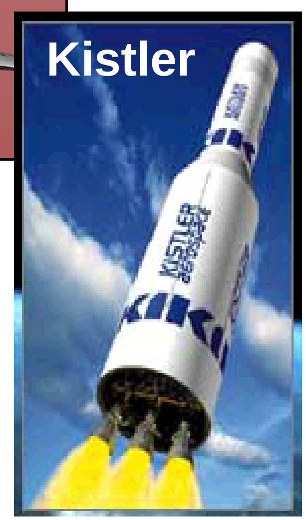
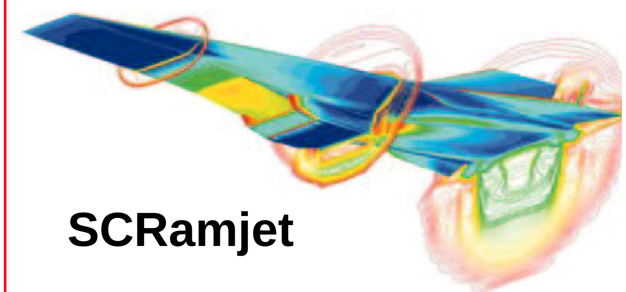
- Gen I





- Gen II



SCRamjet

- Gen III

Space Launch Initiative (SLI)

• **While upgrading the Space Shuttle to keep it flying, 2nd Generation RLV Program activities in the Fiscal Year (FY) 2001 to 2006 timeframe will be directed towards**

-- technical and business risk reduction

-- development of enabling technologies

-- launch vehicle demonstrations

SLI Awards

SLI Partners - Industry

2nd Generation RLV Task Awards NRA 8 - 30 ($1K)
Totals by Company (Base Contracts with Options)

Company	Location	Contract Award	Technology Area
Boeing	Seal Beach, CA	$138,212 $36,412 $74,826 $15,046 $ 6,827 $ 5,101	(Total) TA-1 Systems Studies TA-2 Airframe TA-3 Vehicle Subsystems TA-4 Operations TA-8 Propulsion
Lockheed	Denver, CO	$ 94,319 $36,991 $ 5,226 $25,473 $20,965 $ 4,853 $ 811	(Total) TA-1 Systems Studies TA-2 Airframe TA-3 Vehicle Subsystems TA-4 Operations TA-5 IVHM TA-9 NASA Unique

SLI Awards

Orbital Sciences	Dulles, VA	$ 53,128 $ 5,978 $47,150	(Total) TA-1 Systems Studies [illegible]
Futron	Bethesda, MD	$ 1,856 $ 1,856	(Total) TA-1 System Studies
Northrop/ Grumman	El Segundo, CA	$ 94,341 $ 7,421 $50,455 $36,465	(Total) TA-1 Systems Studies TA-2 Airframe TA-5 IVHM
Oceaneering	Houston, TX	$ 5,347 $ 5,347	(Total) TA-2 Airframe
Materials Research & Design	Rosemont, PA	$ 13,353 $ 13,353	(Total) TA-2 Airframe
Southern Research Institute	Birmingham, AL	$ 1,633 $ 1,633	(Total) TA-2 Airframe
Sierra Lobo	Fremont, OH	$ 4,900 $ 4,900	(Total) TA-4 Operations
PHPK Technologies	Westerville, OH	$ 7,657 $ 7,657	(Total) TA-4 Operations
Honeywell	Glendale, AZ Torrance, CA	$ 11,494 $ 5,044 $ 6,450	(Total) TA-5 IVHM TA-9 NASA Unique

General Kinetics	Lake Forrest, CA	$ 376 $ 376	(Total) TA-6 Upper Stages
Rocketdyne	Canoga Park, CA	$ 65,409 $ 2,747 $62,662	(Total) TA-6 Upper Stages TA-8 Propulsion
MOOG	East Aurora, NY	$ 501 $ 501	(Total) TA-6 Upper Stages
Pratt & Whitney	West Palm Beach, FL	$ 125,817 $ 424 $ 125,393	(Total) TA-6 Upper Stages TA-8 Propulsion
Universal Space Lines	Newport Beach, CA	$ 6,545 $ 6,545	(Total) TA-7 Flight Mechanics
TRW	Redondo Beach, CA	$ 15,544 $ 15,544	(Total) TA-8 Propulsion
Aerojet	Sacramento, CA	$ 7,607 $ 7,607	(Total) TA-8 Propulsion
Andrews Space & Technology	Seattle, WA	$ 3,017 $ 3,017	(Total) TA-8 Propulsion
Kistler	Seattle, WA	$135,400** $ 135,400	(Total) TA- 10 Flight Demonstrations
	(including university contra[illegible]	TOTAL $791,432,000	

SLI Awards

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Space Launch Initiative (SLI)

• Intent is to have at least two competing architectures that will go forward into full-scale development and *could be* operational early next decade (2010 time frame)

• Holy Grail of this program -- Single-Stage-to-orbit (SSTO)

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Why is it So hard to get To orbit in a single stage?

Well... to understand that!
you DO have to be a rocket scientist!

A Quick Refresher on Rocket Theory
Why, What, and How

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Its all About ΔV

- **Compute Required Orbital Speed for 160 km LEO**

$$V_{LEO} = \left[\sqrt{\frac{\mu}{r_{LEO}}}\right] =$$

$$\left[\sqrt{\frac{3.986 \text{ x } 10^{14}\ \dfrac{\text{kg-m}^3}{\text{kg sec}^2}}{[160 \text{ km} + 6371 \text{ km}] \text{ x } 1000\ \dfrac{\text{m}}{\text{km}}}}\right] = 7810\ \frac{\text{m}}{\text{sec}}$$

also ΔV for Polar orbit

Its all About ΔV (cont'd)

- **Compute Earth Rotational velocity at 28.5° (KSC) latitude next (ignore Earth oblateness)**

$$V_{rot_{Earth}} = \omega_{Earth} \times r_{Earth} \times \cos\left[\,Lat\,\right] =$$

$$\left[0.000072921\,\frac{\text{radians}}{\text{sec}}\right] \times \left[6371\ \text{km} \times 1000\,\frac{\text{m}}{\text{km}}\right] \times \cos\left[\frac{28.5\,\pi}{180}\ \text{radians}\right] = 410\,\frac{\text{m}}{\text{sec}}$$

- **For Launch from the cape in to a 28.5° inclination orbit**

$$\Delta V_{\substack{\text{required}\\ \text{for LEO}}} = 7810\,\frac{\text{m}}{\text{sec}} - 410\,\frac{\text{m}}{\text{sec}}\,\frac{\text{m}}{\text{sec}} = 7400\,\frac{\text{m}}{\text{sec}}$$

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Its all About ΔV (cont'd)

- **Compute Earth Rotational velocity at equator (Sea Launch) (ignore Earth oblateness)**

$$V_{rot_{Earth}} = \omega_{Earth} \times r_{Earth} \times \cos[Lat] =$$

$$\left[0.000072921 \frac{radians}{sec}\right] \times \left[6371 \text{ km} \times 1000 \frac{m}{km}\right] \times \cos[0 \text{ radians}] = 465 \frac{m}{sec}$$

$$\Delta V_{\substack{required \\ for\ LEO}} = 7810. \frac{m}{sec} - 465 \frac{m}{sec} \frac{m}{sec} \approx 7350 \frac{m}{sec}$$

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Its all About ΔV (concluded)

- **LEO Launch ΔV's**

$$\begin{bmatrix} \text{Polar orbit: 7810 m/sec} \\ \text{KSC Launch: 7500 m/sec} \\ \text{Equator Launch: 7350 m/sec} \end{bmatrix}$$

- **That ΔV takes a LOT! of Fuel**

How Much Fuel? "The Rocket Equation"

Conservation of momentum leads to the so-called rocket equation, which trades off exhaust velocity with payload fraction. Based on the assumption of short impulses with coast phases between them, it applies to chemical and nuclear-thermal rockets. First derived by Konstantin Tsiolkowsky in 1895 for straight-line rocket motion with constant exhaust velocity, it is also valid for elliptical trajectories with only initial and final impulses.

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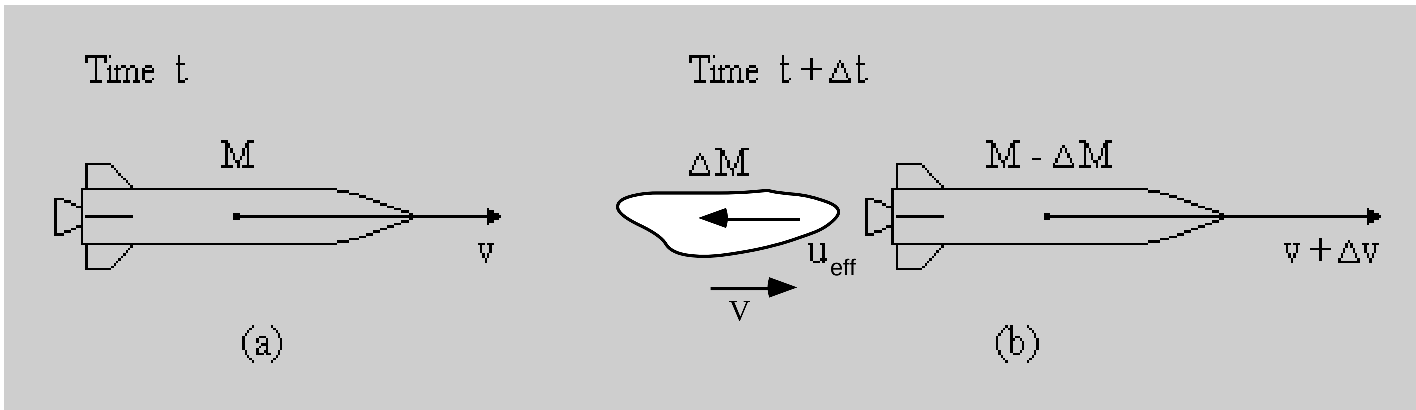
The Rocket Equation

- **You've all seen this derived before**
  .... so here it is:



$$\Delta V_{burn} = g_0 \, I_{sp} \, ln\left[\frac{M_0}{M_0 - \dot{m}_p \Delta t}\right] = g_0 \, I_{sp} \, ln\left[\left(\frac{M_0}{M_{final}}\right)_{burn}\right]$$

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Specific Impulse

$$I_{sp} \equiv \frac{\left|\bar{I}\right|}{m_p} \Rightarrow \boxed{\begin{aligned} \left|\bar{I}\right| &= \textit{total impulse for duration of burn} \\ m_p &= \textit{amount of propellants burned} \end{aligned}}$$

$$\text{Instantaneously: } I_{sp} \equiv \left[\frac{\left| \int_0^t \bar{F}\, dt \right|}{m_p} \right]_{t \to 0} = \frac{\left|\bar{F}\right| dt}{d\, m_p} =$$

$$\frac{\left|\bar{F}\right|}{d\, m_p \,/\, dt} = \boxed{\frac{\left|\bar{F}\right|}{\dot{m}_p}}$$

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Specific Impulse
(cont'd)

- **Historically, I_{sp} was measured in units of *seconds***

$$I_{sp} = \frac{\left|\bar{F}\right|}{\dot{m}_p} \Rightarrow (\text{English Units}) \frac{\cancel{\text{lbf}}}{\cancel{\text{lbm}}/\text{sec}} \approx \text{seconds, right?}$$

Wrong! *lbm*s are not a fundamental unit for mass

(Slugs are the fundamental english unit of mass)

$$I_{sp} = \frac{\left|\bar{F}\right|}{\dot{m}_p} \Rightarrow (\text{MKS units}) \frac{\text{Nt}}{\text{kg/sec}} \approx \frac{\text{kg-m/sec}^2}{\text{kg/sec}} \approx \frac{\text{m}}{\text{sec}}$$

Specific Impulse

(cont'd)

- **Since most engine manufacturers still give $\mathrm{I_{sp}}$ in *seconds* -- we correct for this by letting**

$$\boxed{\mathrm{I_{sp}} \equiv \frac{|\overline{\mathrm{F}}|}{g_0\,\dot{m}_p}} \Rightarrow g_0 \approx 9.81\,\frac{\mathrm{m}}{\mathrm{sec}^2}\ \left[\,\textit{acceleration of gravity at sea level}\,\right]$$

English Units -- use *slugs* not *lbms* !

$$(\text{MKS units})\quad \frac{\dfrac{\mathrm{Nt}}{\mathrm{kg/sec}}}{\dfrac{\mathrm{m}}{\mathrm{sec}^2}} \approx \frac{\dfrac{\mathrm{kg\text{-}m/sec}^2}{\mathrm{kg/sec}}}{\dfrac{\mathrm{m}}{\mathrm{sec}^2}} \approx \mathrm{sec}$$

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Specific Impulse (concluded)

- For chemical Rockets, I_{sp} depends on the type of fuel/oxydizer used

Vacuum I_{SP}

Fuel	Oxidzer	Isp (s)
Liquid propellants		
Hydrogen (LH2)	Oxygen (LOX)	450
Kerosene (RP-4)	Oxygen (LOX)	280
Monomethyl hydrazine	Nitrogen Tetraoxide	310
Solid propellants		
Powered Al	Ammonium Perchlorate	270

- **Most efficient rocket motor ever built, SSME, effective I_{sp} ~ 435 sec**

"Propellant Mass Fraction"

- **How do we compute the amount of propellant required**

$$\frac{M_0}{M_{final}} = \frac{M_{dry} + M_{payload} + M_{\substack{fuel \\ + oxidizer}}}{M_{dry} + M_{payload}} = 1 + P_{mf}$$

$$\Downarrow$$

$$\boxed{P_{mf} \equiv \frac{M_{\substack{fuel \\ + oxidizer}}}{M_{dry} + M_{payload}}}$$

"Propellant Mass Fraction"

Ramifications of the Rocket Equation

- **Substituting** P_{mf} **into the Rocket equation**

$$\boxed{\frac{M_0}{M_{final}} = 1 + P_{mf}}$$

$$\boxed{\Delta V_{burn} = g_0\, I_{sp}\, ln\left[\left(\frac{M_{initial}}{M_{final}}\right)_{burn}\right] = g_0\, I_{sp}\, ln\left[\left(1 + P_{mf}\right)_{burn}\right]}$$

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"Propellant Mass Fraction"

Ramifications of the Rocket Equation (cont'd)

- **Solving for** P_{mf}

$$\left(P_{mf}\right)_{burn} = e^{\left[\frac{\Delta V_{burn}}{g_0 I_{sp}}\right]} - 1$$

- **Mass of Fuel and oxidizer required for a burn to give a specified ΔV**

$$M_{\substack{fuel \\ + oxidizer}} = \left[M_{dry} + M_{payload}\right] \left[e^{\left[\frac{\Delta V_{burn}}{g_0 I_{sp}}\right]} - 1 \right]$$

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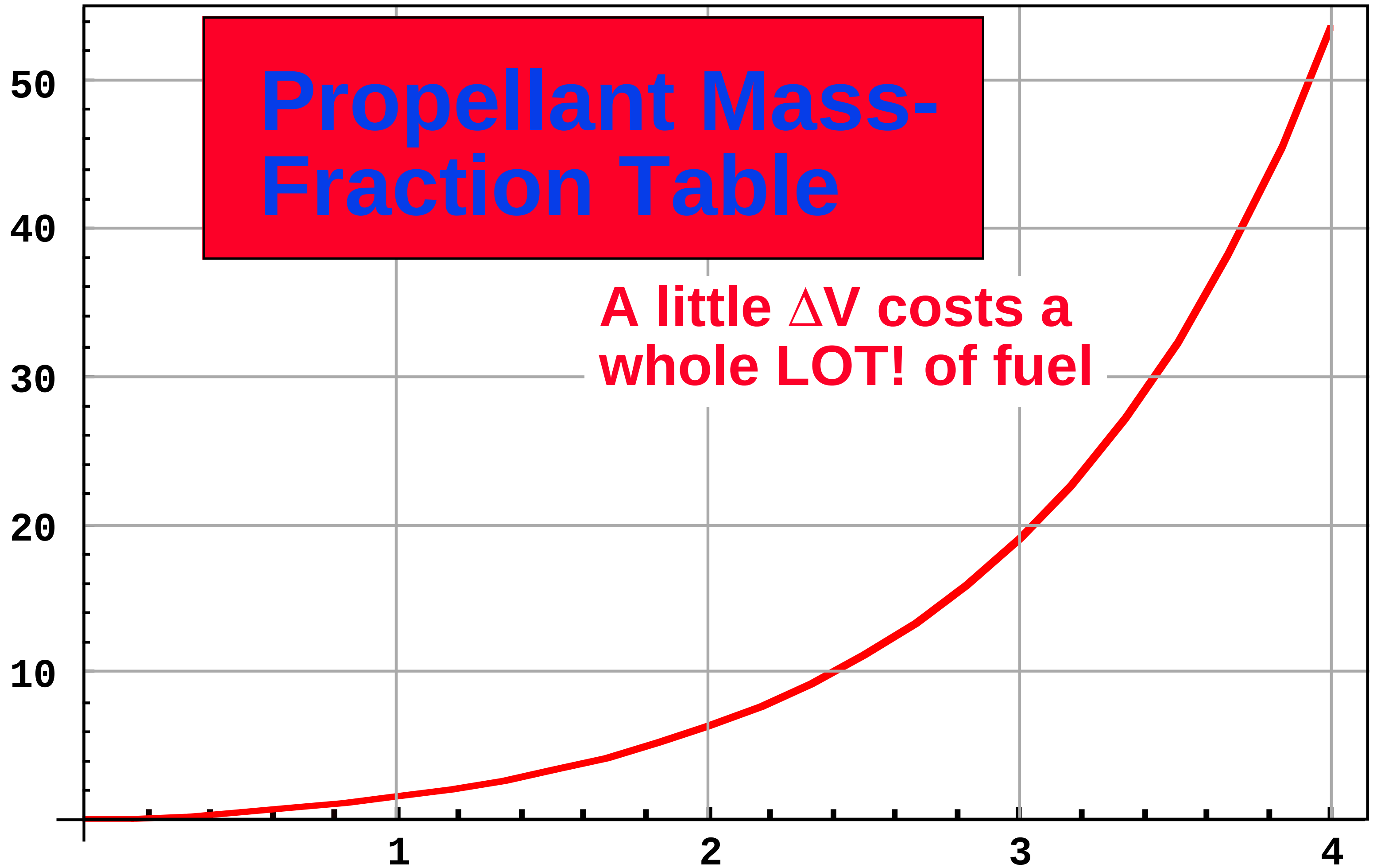
"Propellant Mass Fraction"

Ramifications of the Rocket Equation (concluded)

P_{mf}



$\Delta V_{burn} \, / \, g_0 \, I_{sp}$

Kelly Space & Technology ECLIPSE Vehicle

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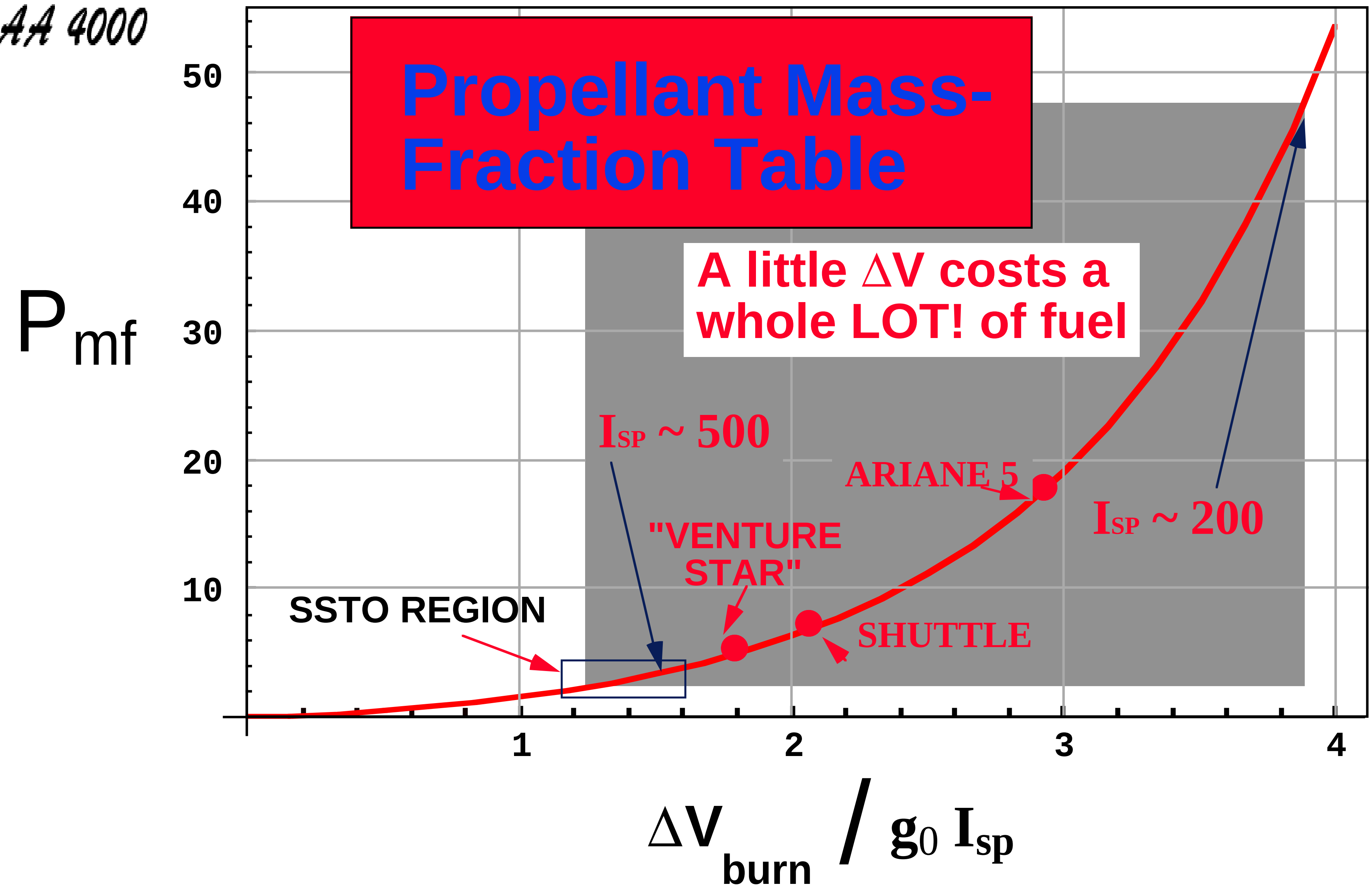
or

a little saved ΔV can save a whole LOT! of fuel

"Propellant Mass Fraction"

Ramifications of the Rocket Equation (concluded)



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Example Calculation:

Propellant mass fraction required for SSTO Ariane 4 Launch from Equator

$$\frac{\Delta V_{burn}}{g_0 \, I_{sp}} = \frac{7348.7 \, \frac{m}{sec}}{\left[9.81 \, \frac{m}{sec^2}\right] \times \left[260 \text{ sec}\right]} = 2.881$$

$$\Downarrow$$

$$\left(P_{mf}\right)_{burn} = e^{\left[\frac{\Delta V_{burn}}{g_0 I_{sp}}\right]} - 1 = e^{[2.881]} - 1 = \boxed{16.84}$$

N204/UH25 (Hypergolic propellants)

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Example Calculation: (concluded)

Propellant mass fraction required for SSTO Ariane 4 Launch from Equator

Ariane 4 with 2 strap on liquid boosters

Strap-On propellant mass: 2 x 87300 lbm = 174600 lbm

Main Booster (stages 1 and 2) propellant mass: 582047 lbm

Gross take-Off weight: 851500 lbm

$$\Rightarrow P_{mf} = \frac{582047 + 174600}{851500 - \left[582047 + 174600\right]} = 7.977$$

- **Aint' No way its going SSTO!**

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How About the Shuttle?

$$P_{mf_{burn}} = \frac{M_{\substack{fuel \\ + oxidizer}}}{M_{dry} + M_{payload}}$$

```
Weight (lb)
 Gross lift-off . . . . . . . .  4,500,000
 External Tank (full) . . . .  1,655,600
 External Tank (Inert) . . . .     66,000
 SRBs (2) each at launch . . . 1,292,000
 SRB inert weight, each . . .  . 192,000
```

$$M_{\substack{fuel \\ + oxidizer}} = \left[M_{\substack{fuel \\ + oxidizer}}\right]_{external\ tank} + \left[M_{\substack{fuel \\ + oxidizer}}\right]_{SRB} =$$

$$[1{,}655{,}600 - 66{,}000] + 2\,[1{,}292{,}000 - 192{,}000] \approx 3{,}789{,}600 \text{ lbs}$$

$$P_{mf_{launch}} = \frac{M_{\substack{fuel \\ + oxidizer}}}{M_{dry} + M_{payload}} = \frac{3{,}789{,}600}{4{,}500{,}000 - 3{,}789{,}600} \approx 5.33$$

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How About the Shuttle? (cont'd)

- **Compute Effective Shuttle Launch I_{sp}**

$$I_{sp}^{(effective)} = \frac{\int F\,dt}{M_{propellant_{burned}}} = \frac{2 \times \left[\int F\,dt\right]_{SRB} + 3 \times \left[\int F\,dt\right]_{SSME}}{2 \times \left[M_{\substack{propellant \\ burned}}\right]_{SRB} + 3 \times \left[M_{\substack{propellant \\ burned}}\right]_{SSME}} =$$

$$\frac{2 \times 2.65 \times 10^6 \text{ lbs} \times 123 \text{ sec}^{(tburn)} + 3 \times 0.454 \times 10^6 \text{ lbs} \times 522 \text{ sec}^{(tburn)}}{\left[2 \times 1{,}100{,}000 \text{ lbm} + 3 \times 1040 \frac{\text{lbm}}{\text{sec}} \times 522 \text{ sec}^{(tburn)}\right]\left[\frac{g_c}{g_0}\right]} =$$

$$\frac{1{,}362{,}864{,}000 \text{ lbf-sec}}{3828640 \text{ lbm}} \frac{\text{lbm-ft / lbf-sec}^2}{\text{ft/sec}^2} = 355.97 \text{ sec}$$

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How About the Shuttle? (concluded)

- **Compute Max availalbe ΔV for Shuttle Launch**

$$\Delta V_{max} = g_0 \, I_{sp} \, ln\left[1 + P_{mf}\right] =$$

$$32.1742 \frac{ft}{sec^2} \times 355.97 \text{ sec} \times ln\left[1 + 5.33\right] =$$

$$21134.32 \frac{ft}{sec} = \boxed{6441.75 \frac{m}{sec} < \underline{7394.7 \frac{m}{sec}} \; Nope!}$$

- **Shuttle ain't getting there either -- that's why they have to dump the solids and the external tank**

$$\boxed{I_{sp} = \underline{\underline{500 \text{ sec}}} \Rightarrow \frac{500 \text{sec}}{355.97 \text{ sec}} \times 6441.75 \text{ m/sec} = 9048.16 \text{ m/sec}}$$

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Ramifications of "the Rocket Equation"

- **Any increase in ΔV must come from increasing I_{sp} or P_{mf}**

  -- First case (I_{sp}) requires adopting a more efficient propulsion system

  --Second case (mass fraction) requires reduction of the structural mass or reduced payload (for same vehicle weight)

  -- Can't just add more propellant -- because that means bigger tanks and the dry weight rises proportionately

- **Reducing payload to obtain more ΔV is a bad-tradeoff**

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Ramifications of "the Rocket Equation" (cont'd)

• **Reducing Structural weight to increase P_{mf} is a viable option** -- but it comes at a high price (adds inherent risks)

-- lighter vehicle tend to damage more easily
-- reduced redundancy in critical sub-systems
-- there are limits as to how light a vehicle can be

• **Best Option is to increase efficiency of the propulsion system (increase I_{sp})**

-- "easier said than done" -- requires significant advances in propulsion technology

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How do we "grow" I_{sp}

Typical I_{sp}'s

Cryogenic:
400 to 440 seconds

Hypergolics:
260 to 290 seconds

Electric (Ion):
2,500-10,000 seconds

Nuclear:
10^2 to 10^3 seconds

Antimatter:
10^7 seconds

If we could just get to here!!!

$$I_{sp} = \underline{\underline{\underline{500\ \text{sec}}}} \Rightarrow$$

$$\frac{500\text{sec}}{355.97\ \text{sec}} \times 6441.75\ \text{m/sec} =$$

$$9048.16\ \text{m/sec}$$

shuttl

Then SSTO is feasible

- **Since nuclear rockets and matter-antimatter engines aren't exactly off-the-shelf technology, and electric propulsions systems produce very low thrust levels, for now we'll just look at the chemical rockets.**

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Specific Impulse (revisited)

- For chemical Rockets, I_{sp} depends on the type of fuel/oxydizer used

Vacuum ISP

Fuel	Oxidizer	Isp (s)
Liquid propellants		
Hydrogen (LH2)	Oxygen (LOX)	450
Kerosene (RP-4)	Oxygen (LOX)	280
Monomethyl hydrazine	Nitrogen Tetraoxide	310
Solid propellants		
Powered Al	Ammonium Perchlorate	270

- **SSME -- Vacuum Isp 452.4, Launch Isp 360, Mean Isp 435**

  -- atmospheric losses kill effectiveness of the rocket engine

- **But is there something we can do with the Nozzle?**

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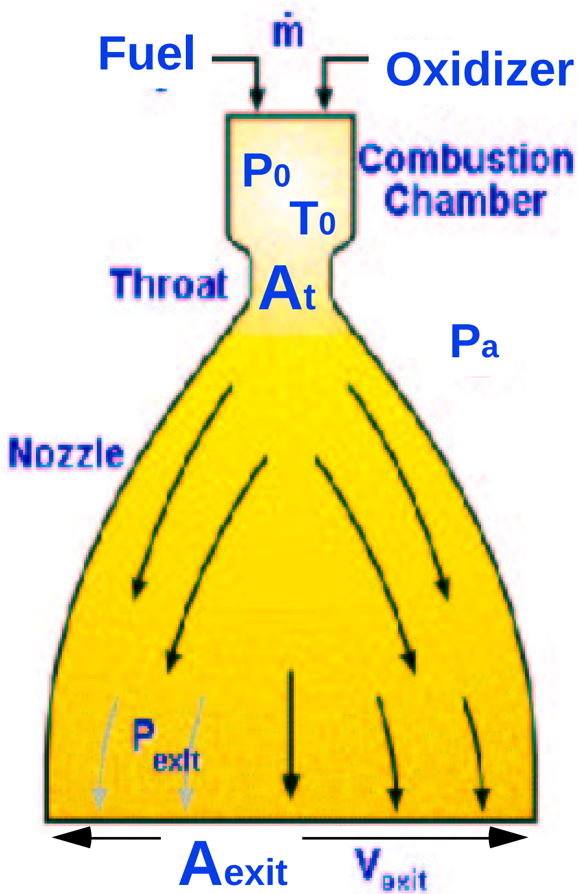
Let's Learn About the Nozzle?

exactly what happens here?



- Propellants combine and burn in combustion chamber
- Combustion products exhaust through throat
- Nozzle expands combustion products, increasing velocity & decreasing pressure

$$T_{hrust} = \dot{m}V_{exit} + A_{exit}\,(P_{exit} - P_{\infty})$$

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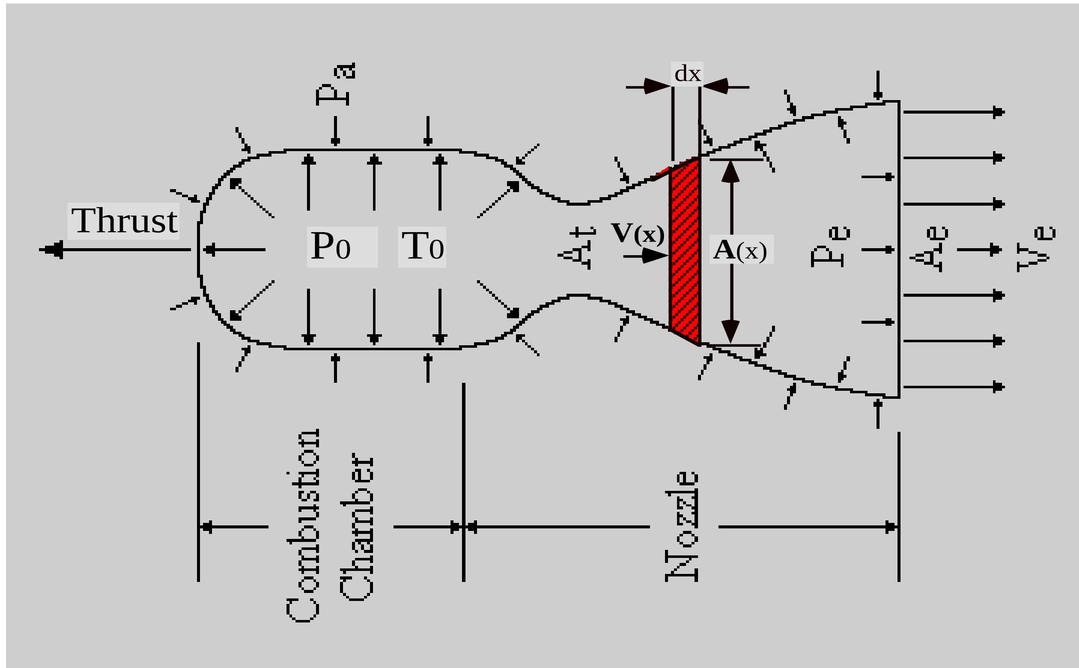
Rocket Nozzle Primer

- **Mass Conservation**



- **Steady Flow: *"continuity equation"***

$$\frac{d[m_x]}{dt} = \left[\frac{d[\rho\, A\, dx]}{dt}\right] = d\left[\rho\, A\, \frac{dx}{dt}\right] = 0 \Rightarrow \boxed{\rho_x\, A_{(x)}\, V_x = \text{constant}}$$

- **log form:**

$$d\left\{ ln\left[\rho\, A\, V = \text{constant}\right]\right\} = d\left\{ln\left[\rho\, A\, V\right]\right\} = 0 \Rightarrow \boxed{\frac{d\rho}{\rho} + \frac{dA}{A} + \frac{dV}{V} = 0}$$

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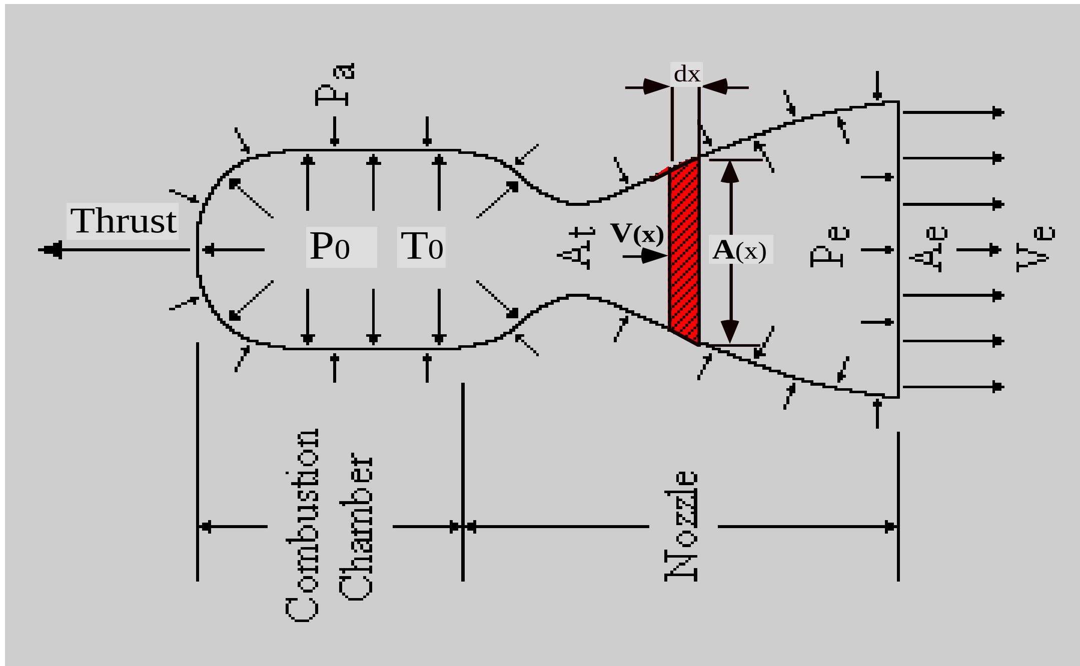
Rocket Nozzle Primer (cont'd)

- **Mass Conservation**



In terms of Mach Number:

Nozzle Equation

$$\frac{dV}{V} = \left[M^2 - 1\right]\frac{dA}{A}$$

$$M = \frac{V}{c}$$

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Ramifications of Continuity Equation

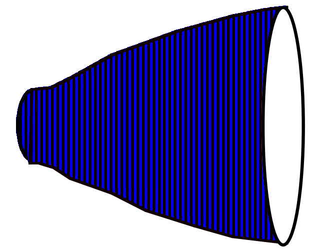
Nozzle Equation

$$\frac{dV}{V} = \left[M^2 - 1\right]\frac{dA}{A}$$

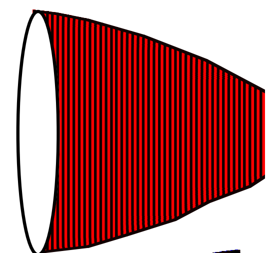
- **Amazing!**

Subsonic: M < 1

$$\frac{dA}{A} > 0 \qquad \frac{dV}{V} < 0 \quad \text{(velocity decreases)}$$

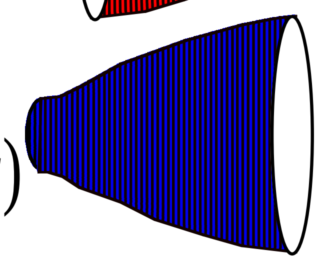


$$\frac{dA}{A} < 0 \qquad \frac{dV}{V} > 0 \quad \text{(velocity increases)}$$

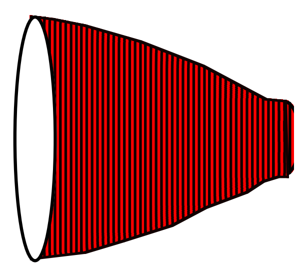


Supersonic: M > 1

$$\frac{dA}{A} > 0 \qquad \frac{dV}{V} > 0 \quad \text{(velocity } \textit{increases}\text{)}$$



$$\frac{dA}{A} < 0 \qquad \frac{dV}{V} < 0 \quad \text{(velocity decreases)}$$



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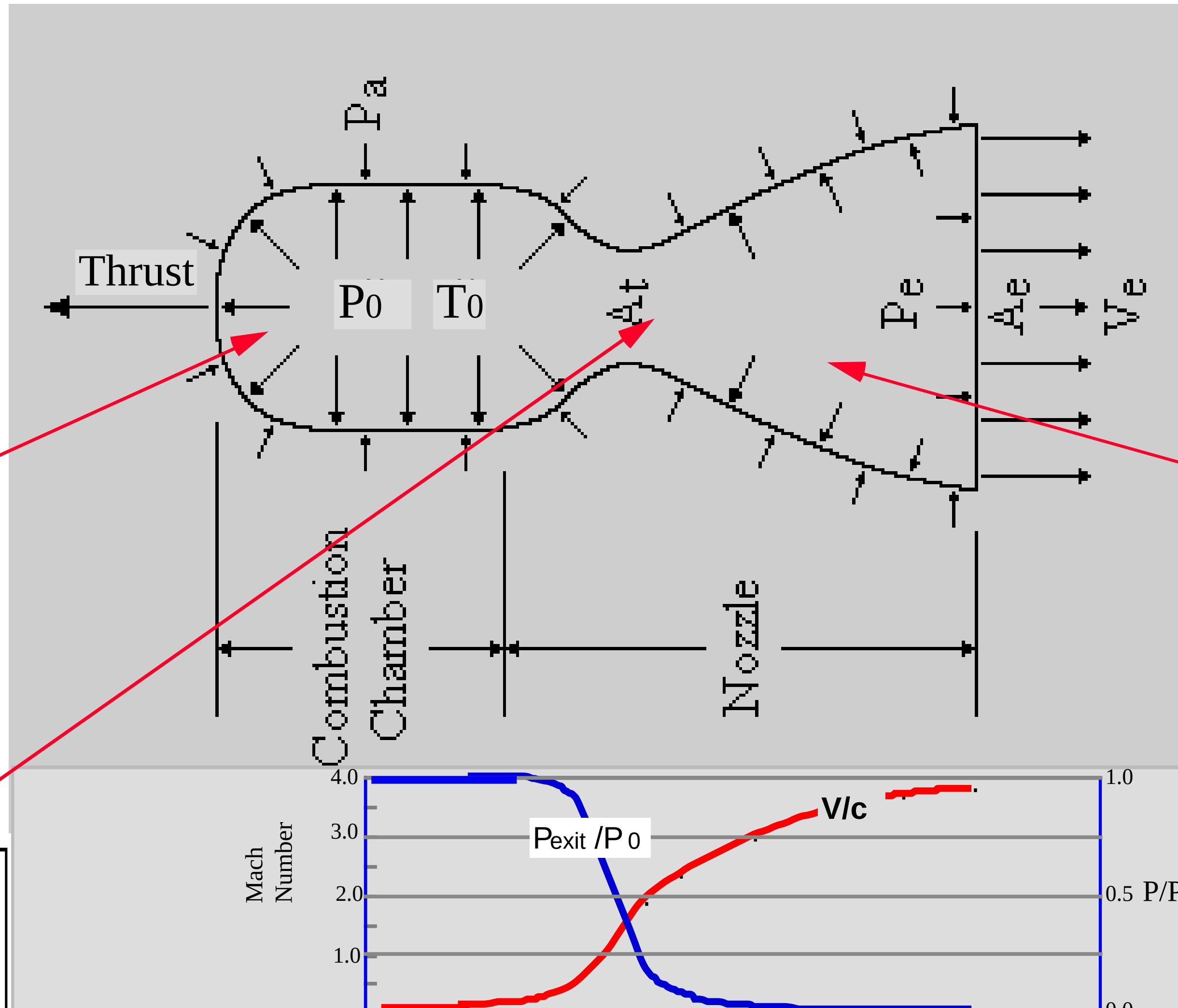
Rocket Nozzle Design Rules



M<1

M=1

M>1

$$\left[\frac{dA}{A}\right]_{throat} \equiv 0$$

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Condition for choked flow:

- **maximum mass flow you can shove through a nozzle**

$$\left[\frac{\dot{m}_{max}}{A_{throat}}\right] = \frac{P_0}{\sqrt{T_0}}\sqrt{\frac{\gamma}{R_g}\left[\frac{2}{\gamma+1}\right]^{\frac{\gamma+1}{\gamma-1}}}$$

Maximum Mass Flow is Dependant on Propellant Combustion Characteristics

$$\boxed{\left[\frac{\dot{m}_{max}}{A_{throat}}\right] = F\left[P_0, T_0, \gamma\right]}$$

Nozzle and Burner Materials limit what is achievable

SS/AA 4000

Now let's Revisit I_{sp}

$$I_{sp} = \frac{T_{hrust}}{\dot{m}} = \frac{1}{g_c}\left[V_{exit} + \frac{\left[p_{exit} - p_a \right]}{\dot{m}} A_e \right] =$$

$$\frac{1}{g_c}\left[V_{exit} + \frac{\left[p_{exit} - p_a \right]}{\dot{m}/ A_{throat}} \frac{A_e}{A_{throat}} \right]$$

- **And *after a miracle occurs*, we get the result**

$$I_{sp} = \frac{\sqrt{R_g T_0}}{g_c} \sqrt{\left(\frac{2\gamma}{\gamma-1}\right)\left[1 - \left(\frac{p_{exit}}{P_0}\right)^{\frac{\gamma-1}{\gamma}}\right]} \left[1 + \frac{\left[\frac{p_{exit}}{P_0} - \frac{p_a}{P_0}\right]}{\left(\frac{p_{exit}}{P_0}\right)^{\frac{1}{\gamma}} \left(\frac{2\gamma}{\gamma-1}\right) \left[1 - \left(\frac{p_{exit}}{P_0}\right)^{\frac{\gamma-1}{\gamma}}\right]} \right]$$

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I_{SP} (whew!)

$$I_{sp} = \frac{\sqrt{R_g T_0}}{g_c} \sqrt{\left(\frac{2\gamma}{\gamma-1}\right)\left[1 - \left(\frac{p_{exit}}{P_0}\right)^{\frac{\gamma-1}{\gamma}}\right]} \left[1 + \frac{\left[\frac{p_{exit}}{P_0} - \frac{p_a}{P_0}\right]}{\left(\frac{p_{exit}}{P_0}\right)^{\frac{1}{\gamma}} \left(\frac{2\gamma}{\gamma-1}\right)\left[1 - \left(\frac{p_{exit}}{P_0}\right)^{\frac{\gamma-1}{\gamma}}\right]} \right]$$

Function of Propellant Combustion Chemistry

$$\boxed{\left[A_{exit}, p_{exit}, p_a\right] \Rightarrow \text{free parameters}}$$

$$\frac{A_{throat}}{A_{exit}} = \sqrt{\left(\frac{\gamma+1}{2}\right)^{\frac{(\gamma+1)}{(\gamma-1)}} \frac{2}{\gamma-1}} \left[\frac{p_{exit}}{P_0}\right]^{\frac{1}{\gamma}} \sqrt{\left[1 - \left(\frac{p_{exit}}{P_0}\right)^{\frac{\gamma-1}{\gamma}}\right]}$$

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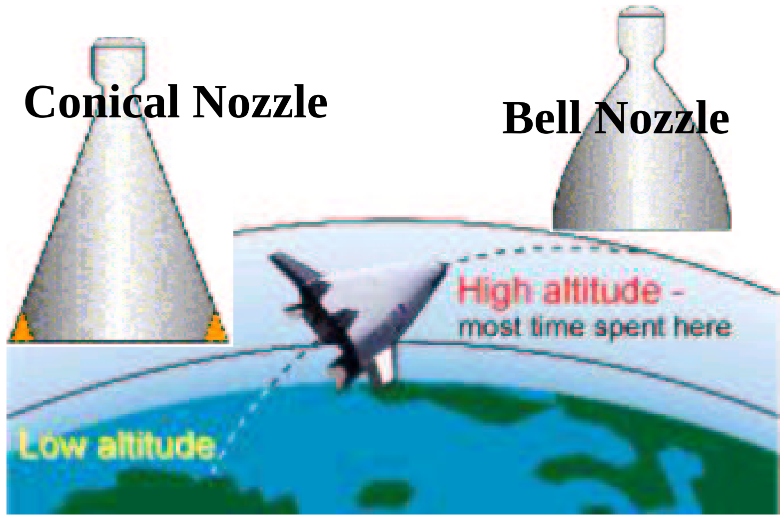
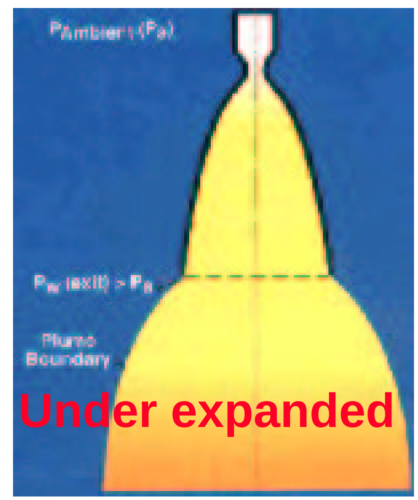
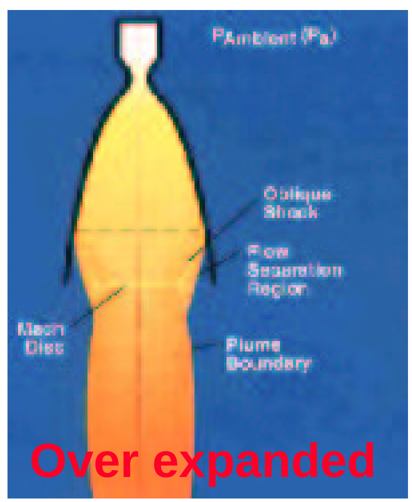
Exit Pressure has a dramatic effect on Nozzle performance



Vacuum (Space)



Lift off



Large area ratio nozzles at sea level cause flow separation, performance losses, high nozzle structural loads

Bell constrains flow limiting performance

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Lets Look at an SSME Example

Thrust data

Sea level Thrust (lbf)
375000.0

Vacuum Thrust (lbf)
470000.0

Vacuum Isp
452.50

Combustor Data

Po (psia)
3125.000

To (deg F)
6144.350

Area values

Aexit (ft^2)
44.89128

A* (ft^2)
0.57922

Ae/A*
0.01290

A*/Ae
77.50284

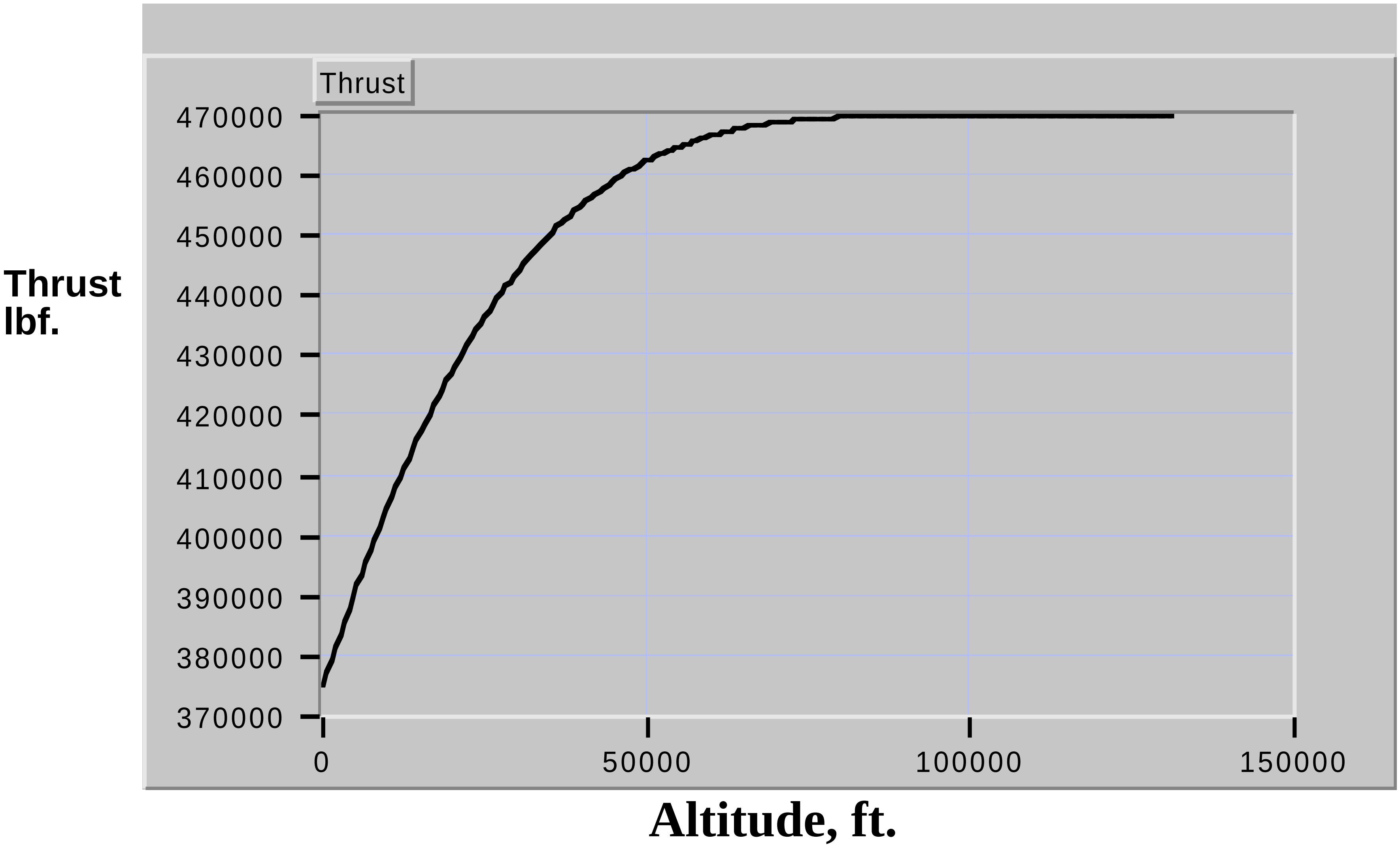
Pexit (psf)
407.30

Exit mach
4.70666

Exit Velocity
13992.449

SS/AA 4000

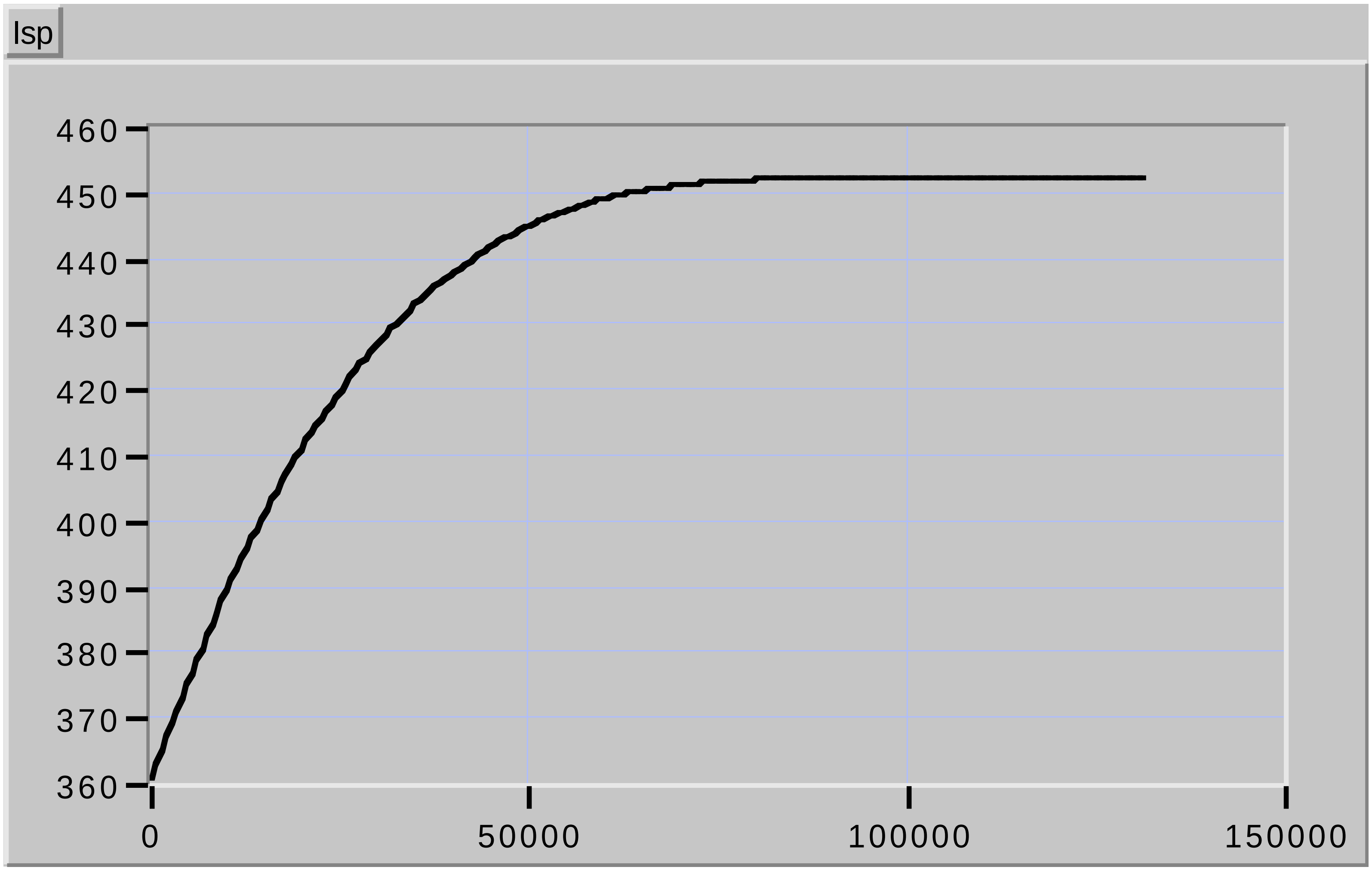
SSME Thrust (lbf) vs Altitude (ft.)



SS/AA 4000

SSME I_{sp} (sec) vs Altitude (ft.)

I_{sp}
sec.



Altitude, ft.

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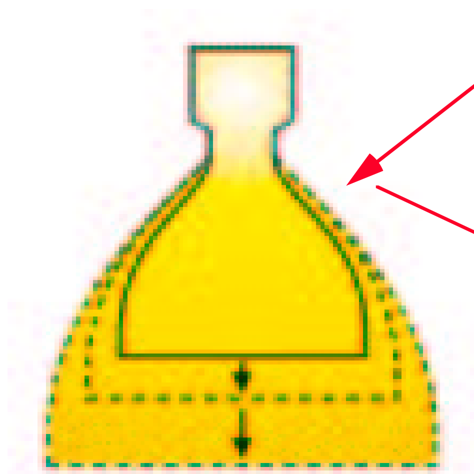
The "Optimum Nozzle"

- **Expanding nozzle increases V_{exit}, but decreases P_{exit} -- there is trade-off here**

- **It can be shown using variational calculus on the relationships from the previous pages that the Optimum nozzle performance occurs when**

$$\frac{A_{exit}}{A_t} \Rightarrow p_{exit} = p_a$$



"telescoping nozzle"

Unfeasible because of the large weight penalty and complexity of deployment mechanisms, also requires that nozzle expand to very large area ratios

"Optimum Nozzle" -- but what would we gain?

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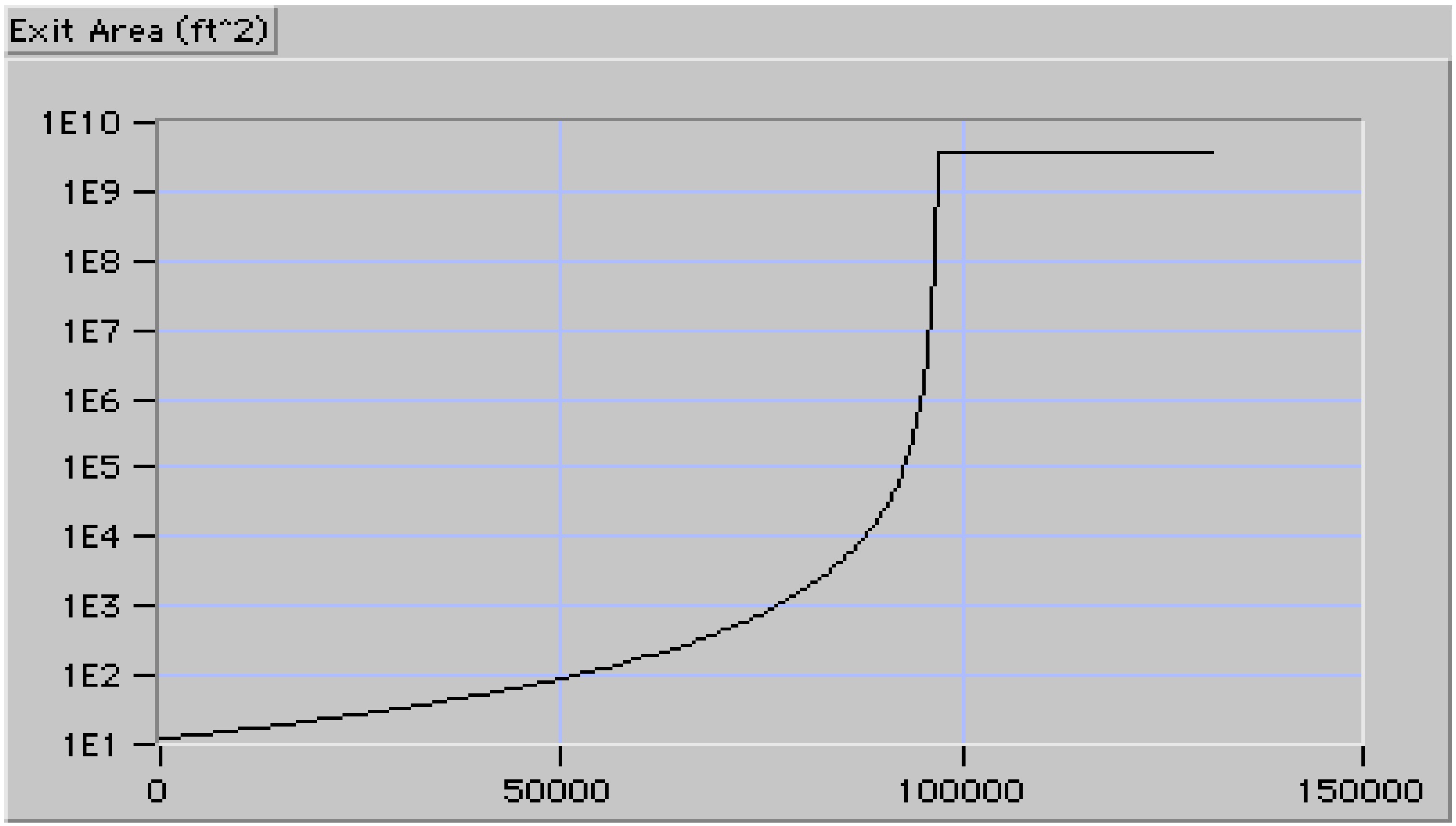
- Let's re-visit the SSME, But this time we allow the nozzle to expand so that P_{exit} tracks $P_{ambient}$



A_{exit}, ft^2

altitude , ft

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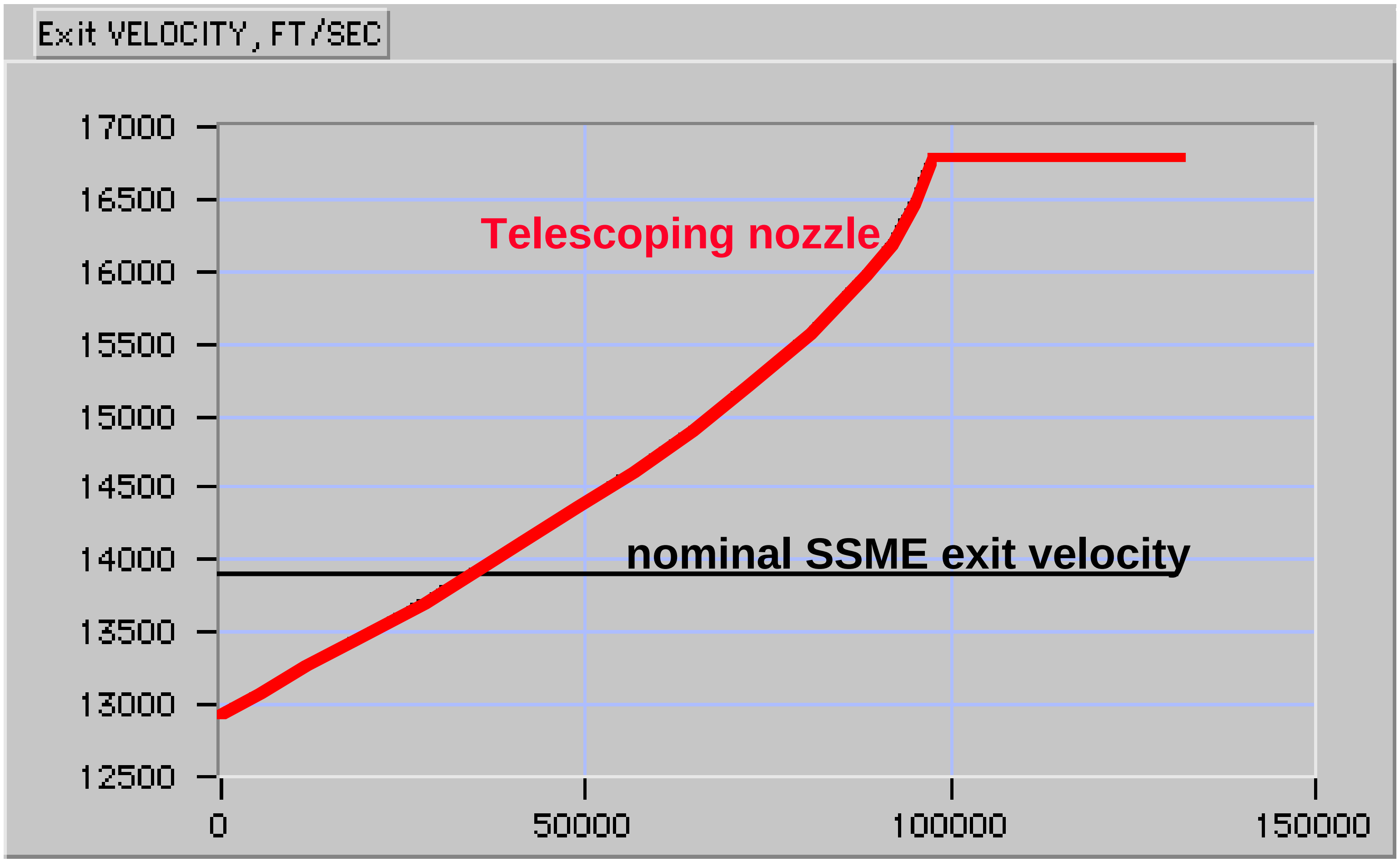
"Optimum Nozzle" (cont'd)

- **Exit Velocity**

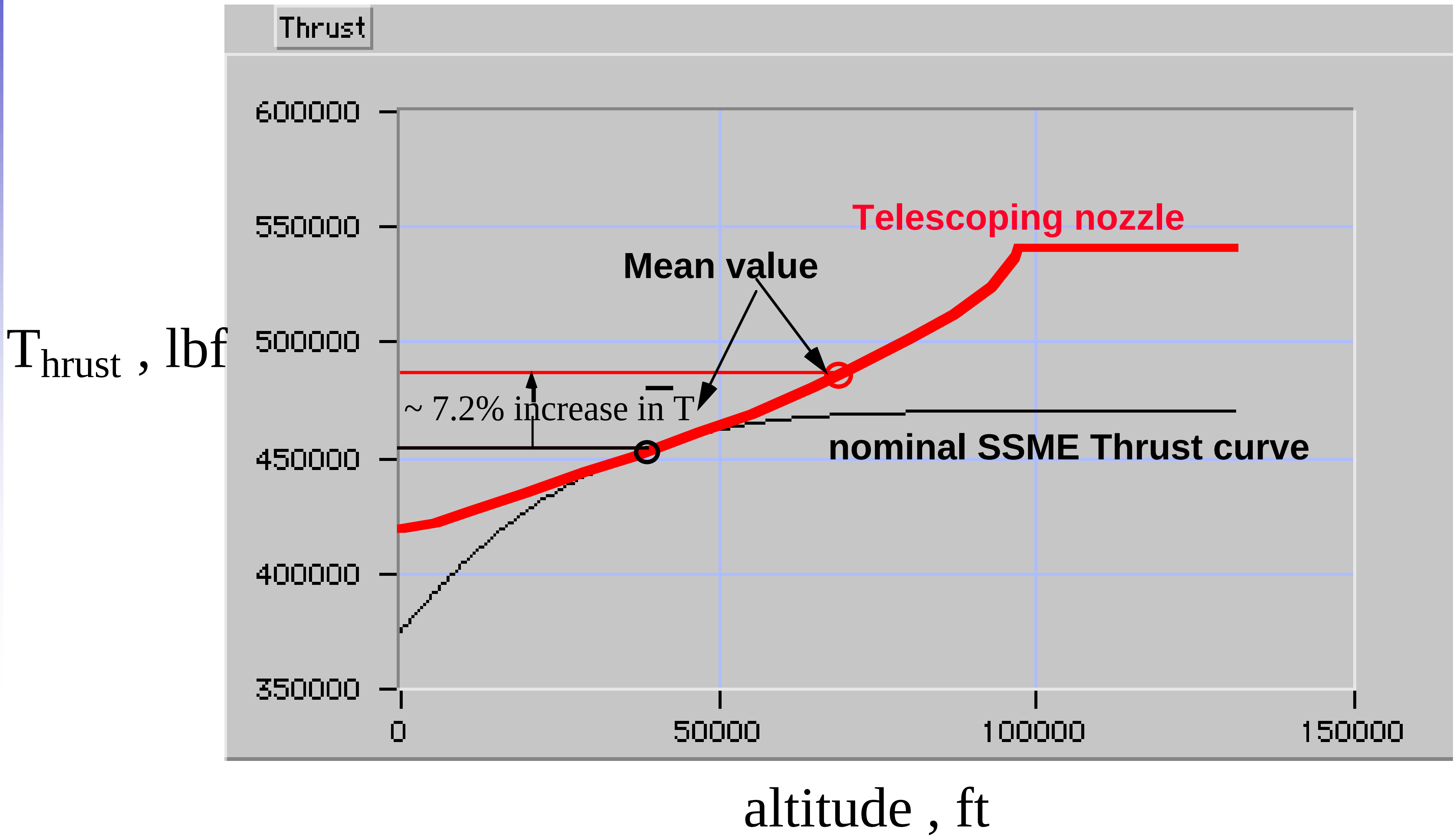
V_{exit} , $\frac{ft}{sec}$



altitude , ft

"Optimum Nozzle" (cont'd)

- **Thrust**



"Optimum Nozzle" (concluded)

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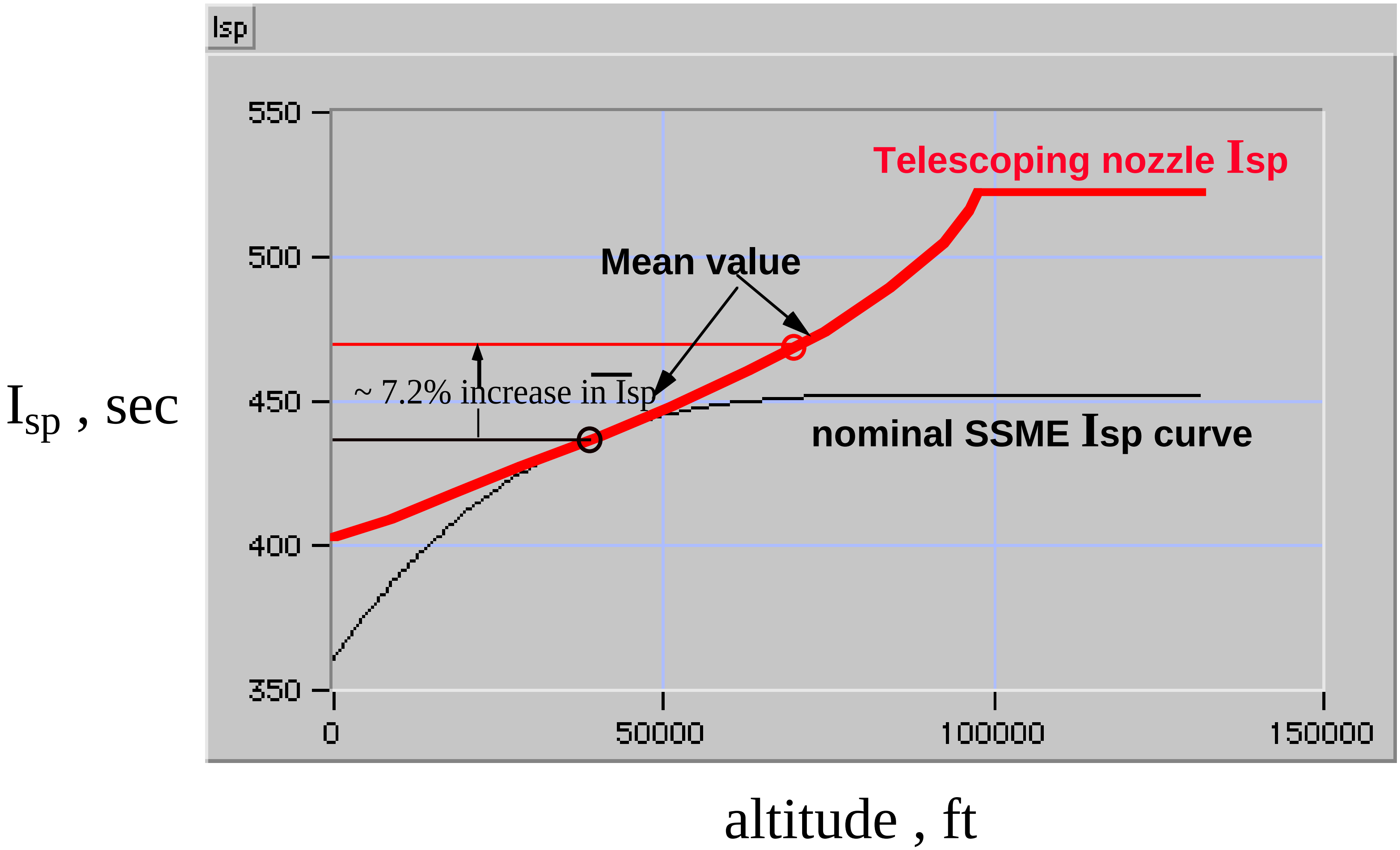
- Isp



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What Would the New I_{sp} be?

$$I_{sp}^{(effective)} = \frac{\int F\,dt}{M_{propellant_{burned}}} = \frac{2\times\left[\int F\,dt\right]_{SRB} + 3\times \underline{1.072}\times\left[\int F\,dt\right]_{SSME}}{2\times\left[M_{propellant\ burned}\right]_{SRB} + 3\times\left[M_{propellant\ burned}\right]_{SSME}} =$$

$$\frac{2\times 2.65\times 10^{6}\text{ lbs}\times 123\text{ sec}^{(tburn)} + 3\times \underline{1.072}\times 0.454\times 10^{6}\text{ lbs}\times 522\text{ sec}^{(tburn)}}{\left[2\times 1{,}100{,}000\text{ lbm} + 3\times 1040\,\frac{\text{lbm}}{\text{sec}}\times 522\text{ sec}^{(tburn)}\right]\left[\frac{g_c}{g_0}\right]} =$$

$$\frac{1{,}415{,}513{,}472\text{ lbf-sec}}{3828640\text{ lbm}}\;\frac{\text{lbm-ft / lbf-}\cancel{\text{sec}}^{2}}{\text{ft/sec}^{2}} = \underline{369.72\text{ sec}}$$

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What Would the New ΔV be?

- **Compute Max availalbe ΔV for Shuttle Launch**

$$\Delta V_{max} = g_0 I_{sp} \, ln\left[1 + P_{mf}\right] =$$

$$32.1742 \frac{ft}{sec^2} \times 369.72 \, sec \times ln\left[1 + 5.33\right] =$$

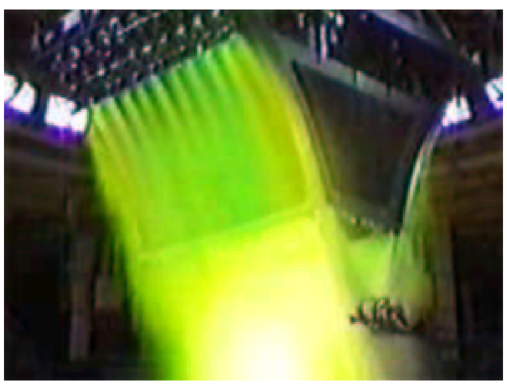
$$21950.67 \frac{ft}{sec} = \boxed{6690.5 \frac{m}{sec} < \underline{7394.7 \frac{m}{sec}} \; \textit{Nope! But better!}}$$

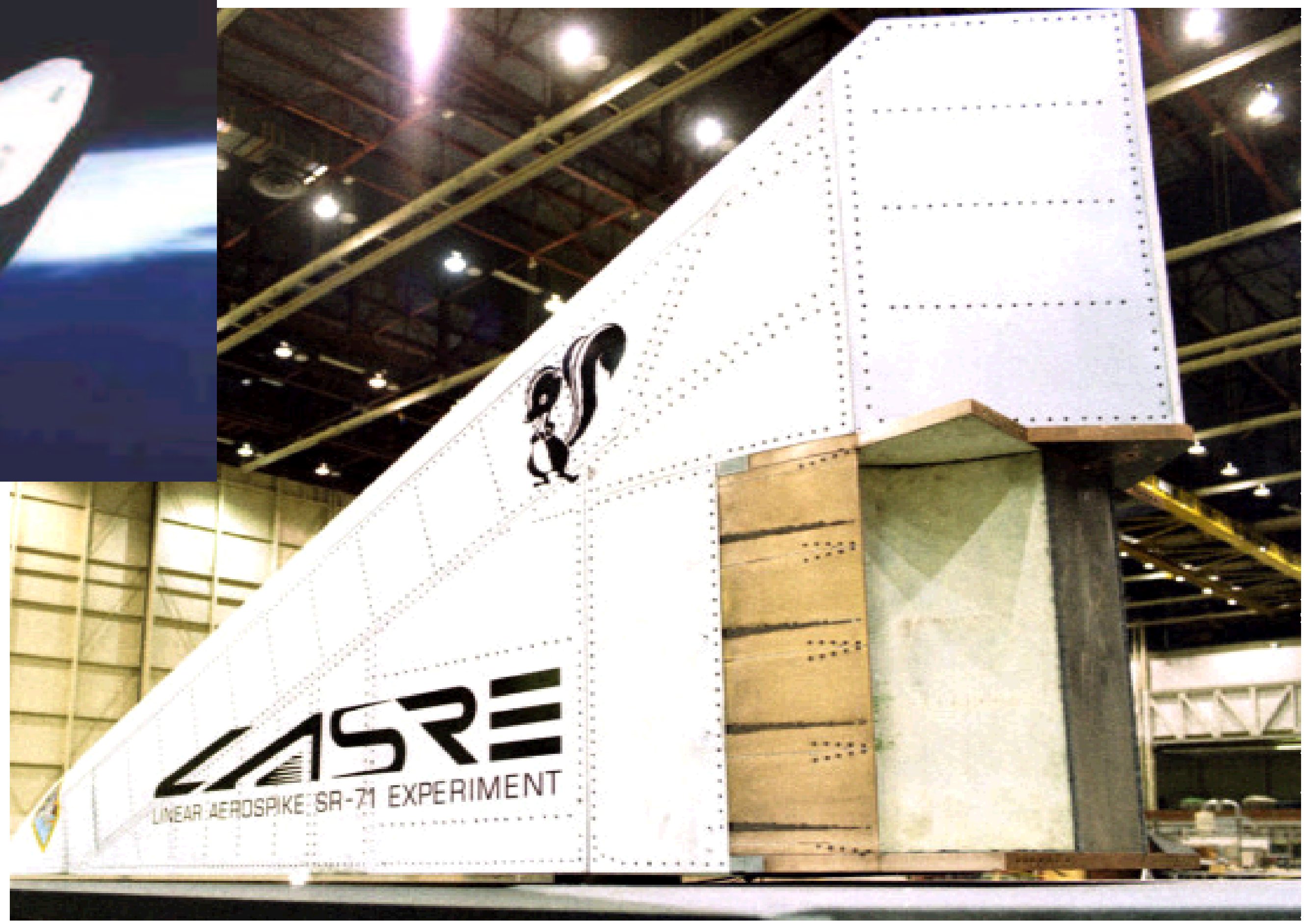
- **Still gotta find a way to lose the solids**

$$\boxed{I_{sp} = \underline{\underline{465.5 \text{ sec}}} \Rightarrow \frac{465.5\text{sec}}{355.97 \text{ sec}} \times 6441.75 \text{ m/sec} = 8423.84 \text{ m/sec}}$$

SS/AA 4000

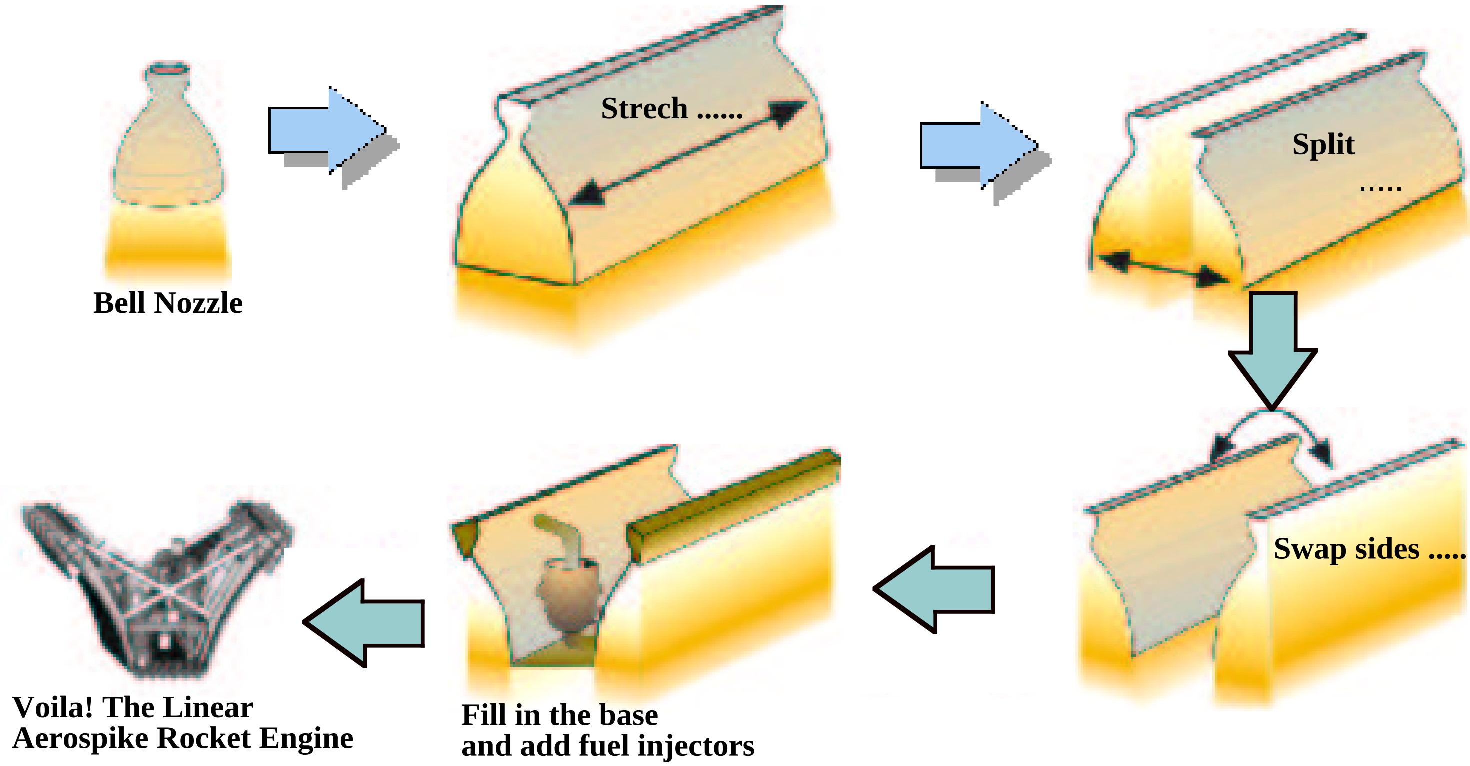
"The Linear Aerospike Rocket Engine"

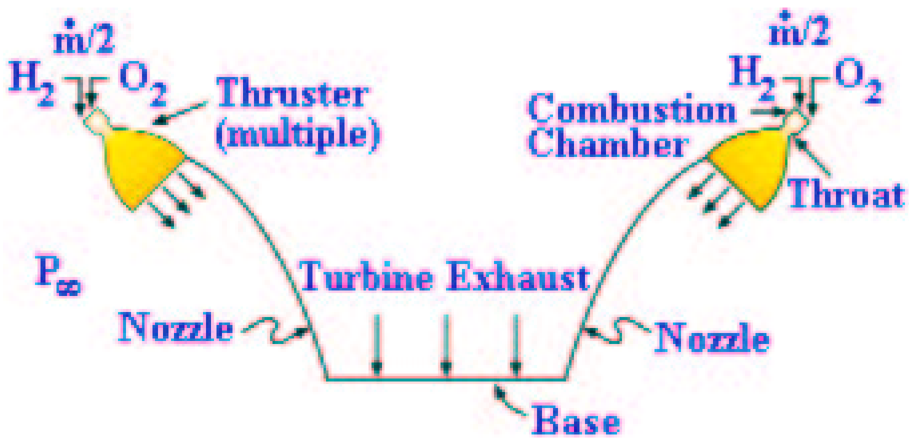




SS/AA 4000 Lift off

A New Nozzle Shape



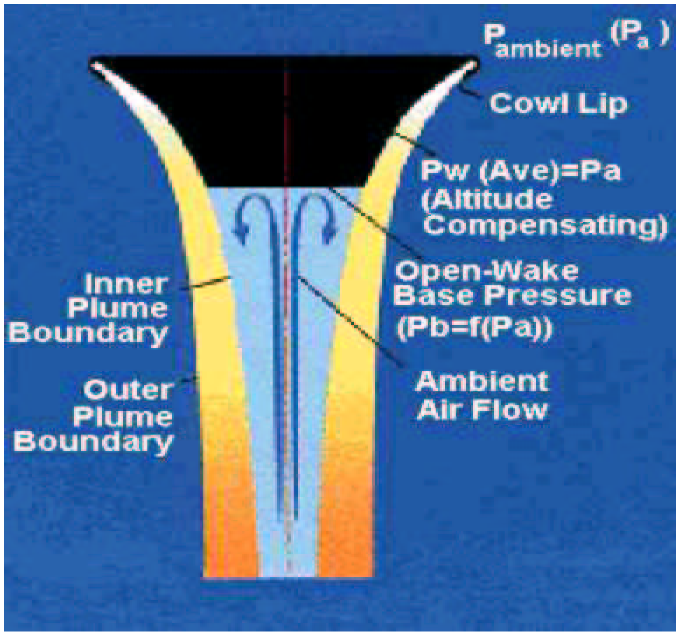


SS/AA 4000

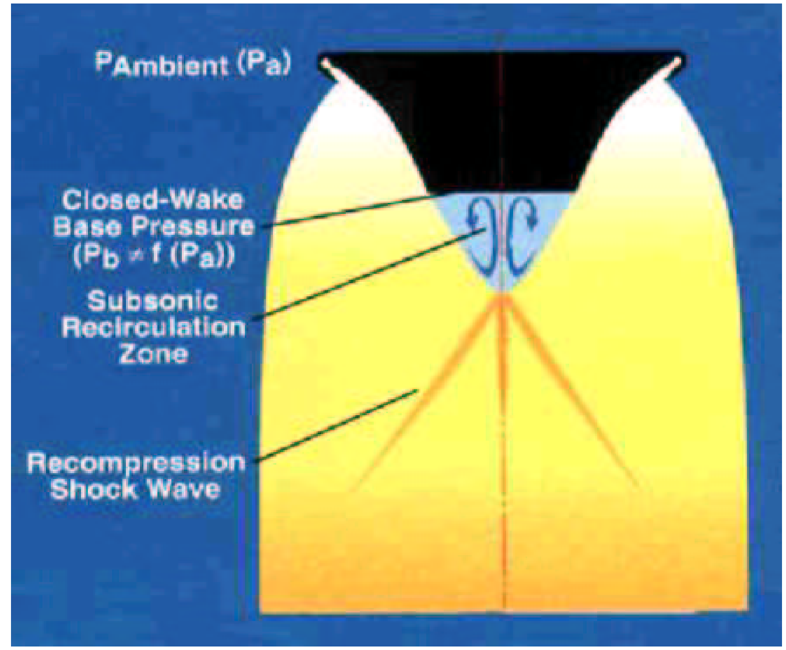
Linear Aerospike Rocket Engine

Nozzle has same effect as telescope nozzle

Lift off



Vacuum (Space)



$$F = F_{Thruster} + F_{Ramp} + F_{Base}$$

$$F_{Thruster} = \cos\theta\,(\dot{m}V_{exit} + A_{exit}(P_{exit} - P_{\infty}))$$

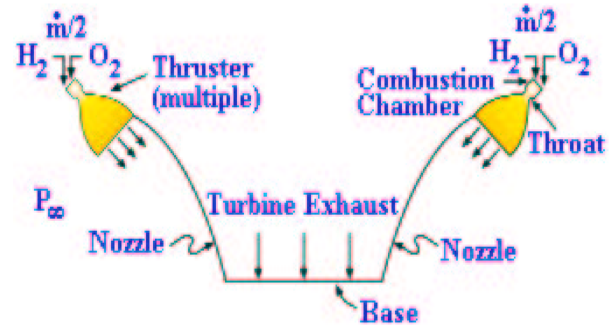
$$F_{Ramp} = \int^{A_{Ramp}} (P_{Ramp} - P_{\infty})\, dA$$

$$F_{Base} = A_{Base}\,(P_{Base} - P_{\infty})$$

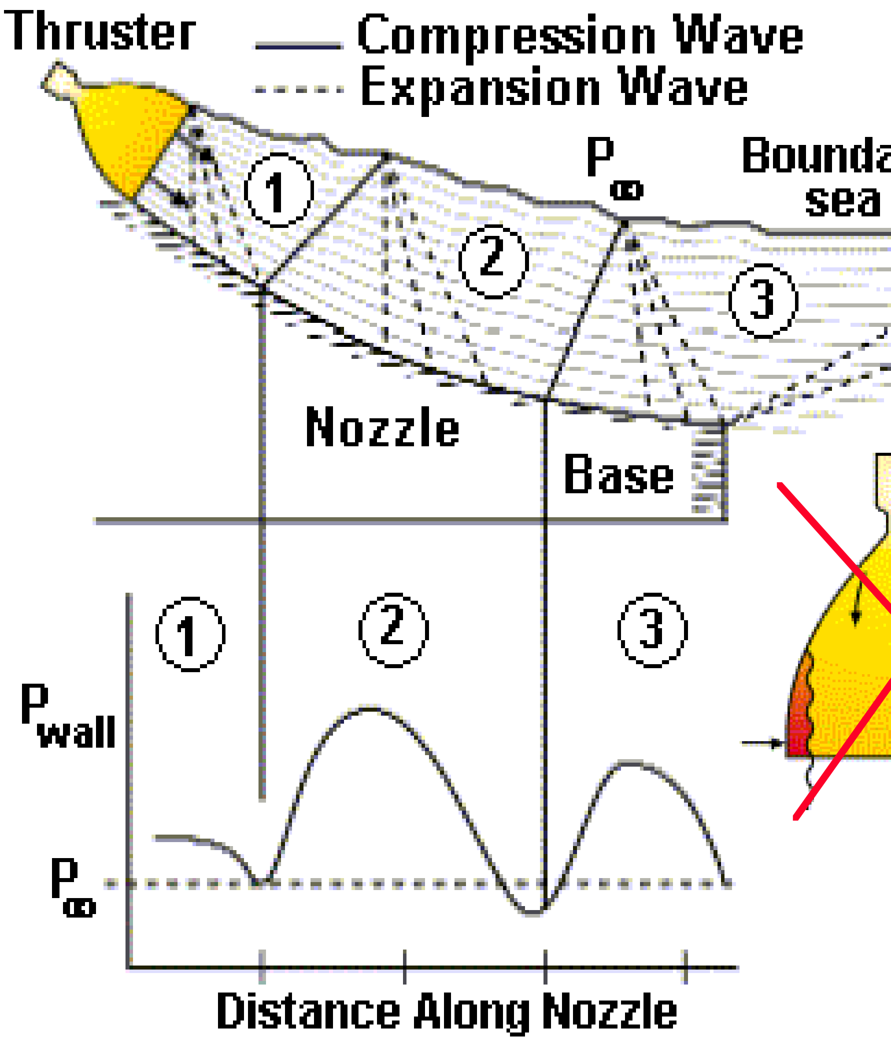
- Aerospike's flow unconstrained, allows best performance

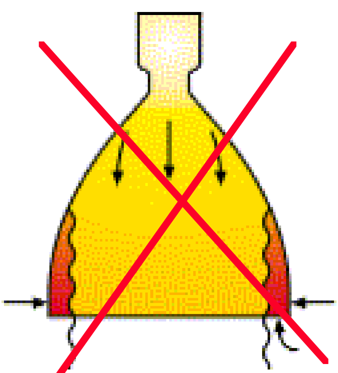
SS/AA 4000

Linear Aerospike Rocket Engine
(cont'd)



Low Altitude Aerodynamics





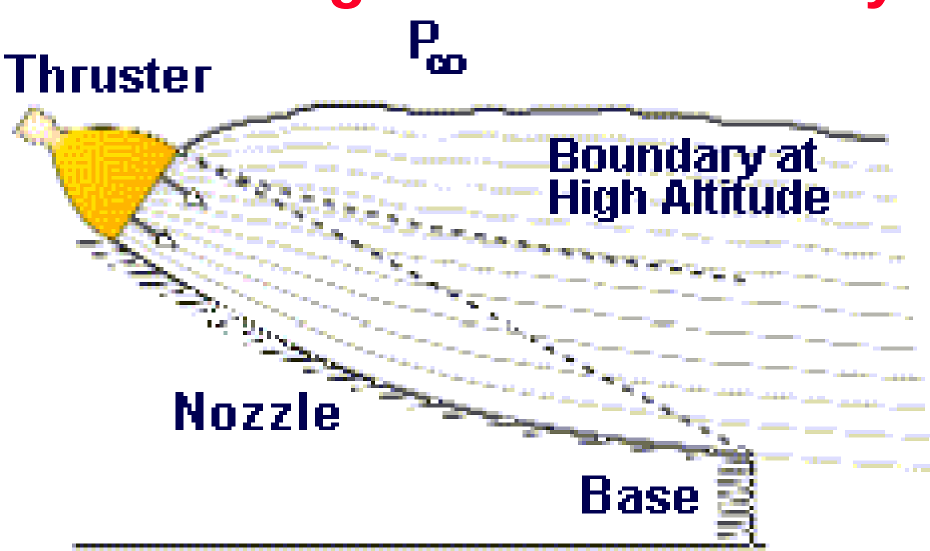
-
- Thruster flow discharges to ramp
- Expansion waves turn flow axially
- Ramp curves, turns flow axially (at low altitudes)
- Turning causes compression wave from (1) to (2) - nozzle pressure increases
- Compression wave reflects off boundary causing expansion waves
- Flow crosses expansion waves in (2) - nozzle pressure decreases
- Ramp continues to curve and turn flow
- Process repeats (2) to (3)

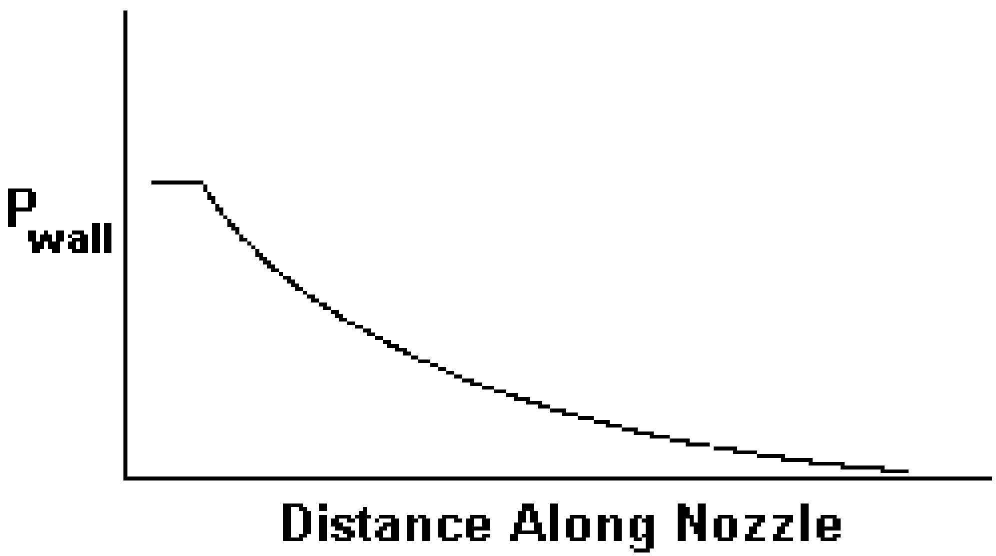
Average nozzle pressure > P, therefore no losses or separation, therefore large area ratio nozzle can be used, enabling SSTO

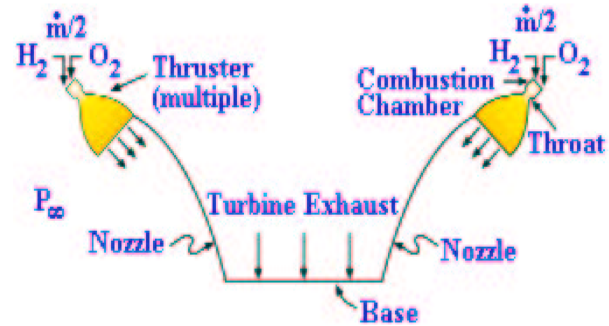
SS/AA 4000

Linear Aerospike Rocket Engine
(cont'd)

High Altitude Aerodynamics







- Thruster flow discharges to ramp
- Expansion waves turn flow axially
- No compression waves exist - all flow turning done by expansion waves
- Nozzle behaves likes a bell

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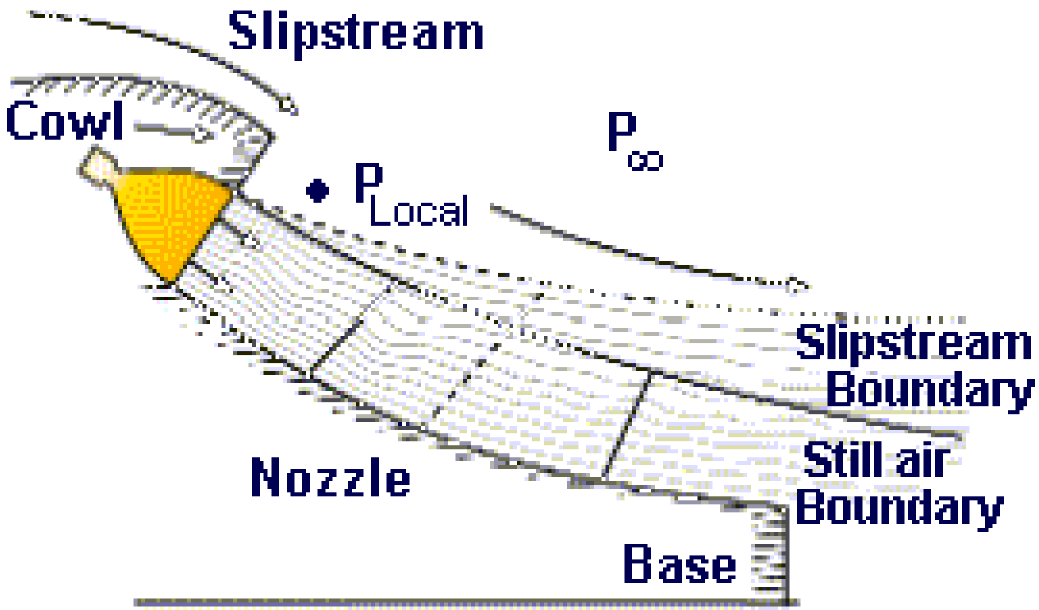
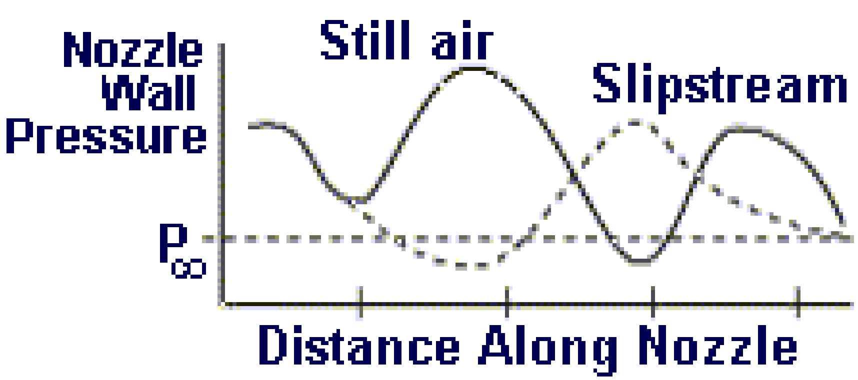
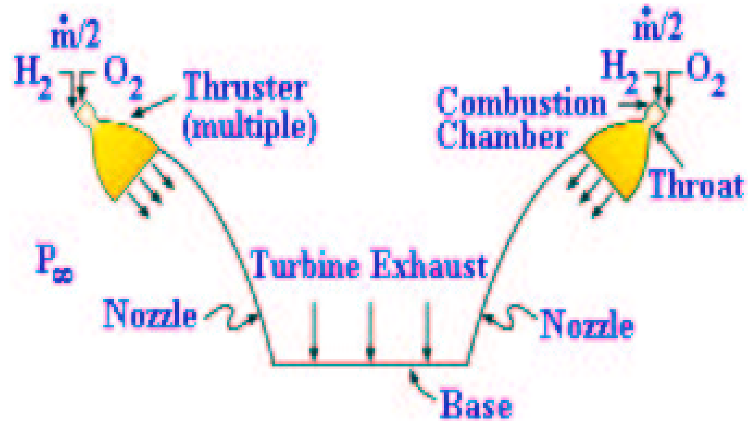
Linear Aerospike Rocket Engine
(concluded)

SlipStream effects







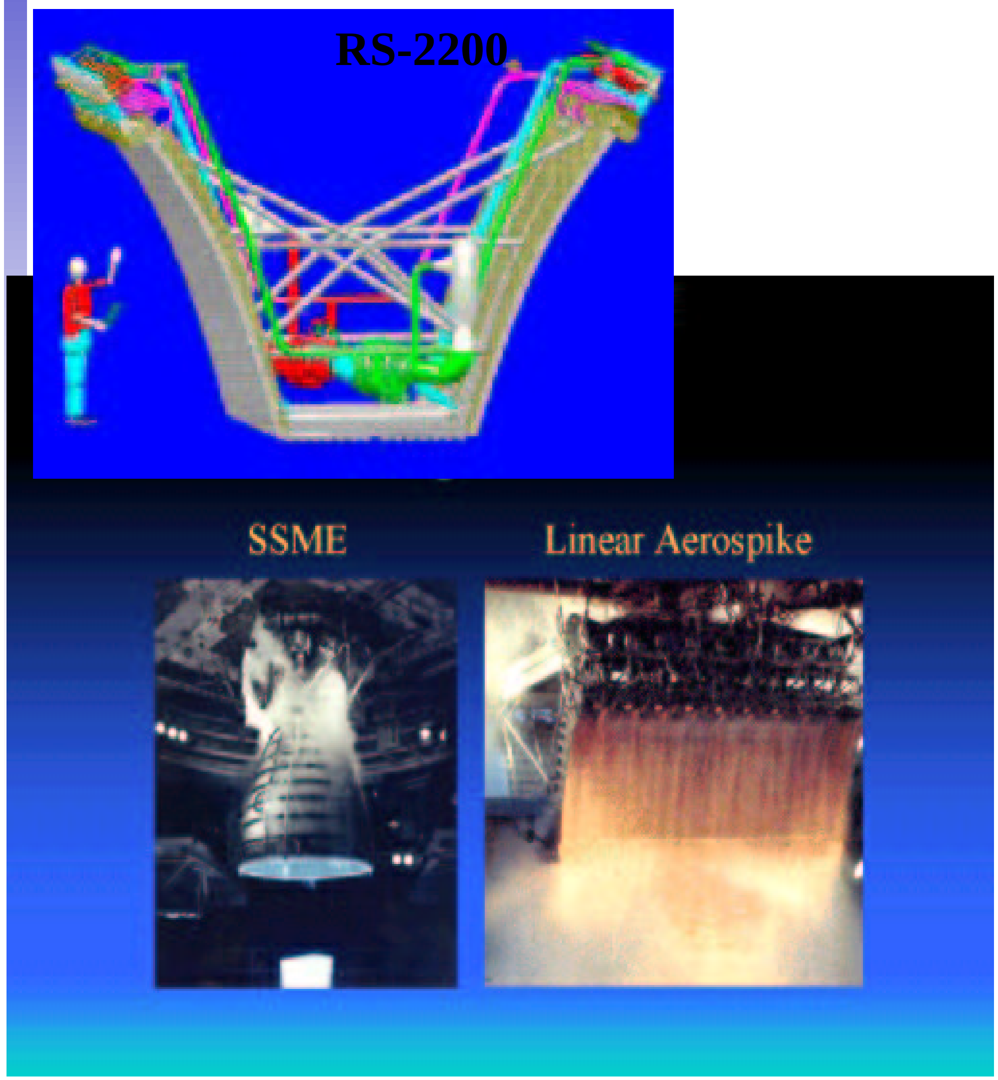
- Air streaming over cowl lowers local pressure - $P_{Local} < P_{infinity}$
- Exhaust plume expands beyond still air case
- Expansion and compression wave systems move aft from still air case

- Resulting recompression Delays Nozzle separation

Bottom Line is that the Linear Aerospike engine realizes about 50% of thetheoretical Isp gains offeredby the Telescoping nozzle

SS|AA 4000

Linear Aerospike Engine Comparison to SSME



RS-2200: (Venture-Star)
Manufacturer: Boeing Rocketdyne
Weight: 8000 lbs.
Max Thrust: 520,000 lbf (Liftoff)
564,000 lbf (Space)
I_{sp}: 420 sec (Liftoff)
460 sec (Space)

Mean Isp: 453.3

SSME: (Shuttle (Block IIa)
Manufacturer: Boeing Rocketdyne
Weight: 7,480 lbs.
Max Thrust: 418,660 lbf (Liftoff)
512,950 lbf (Space)
I_{sp}: 360 sec (Liftoff)
452.4 sec (Space)

Mean Isp: 437.0

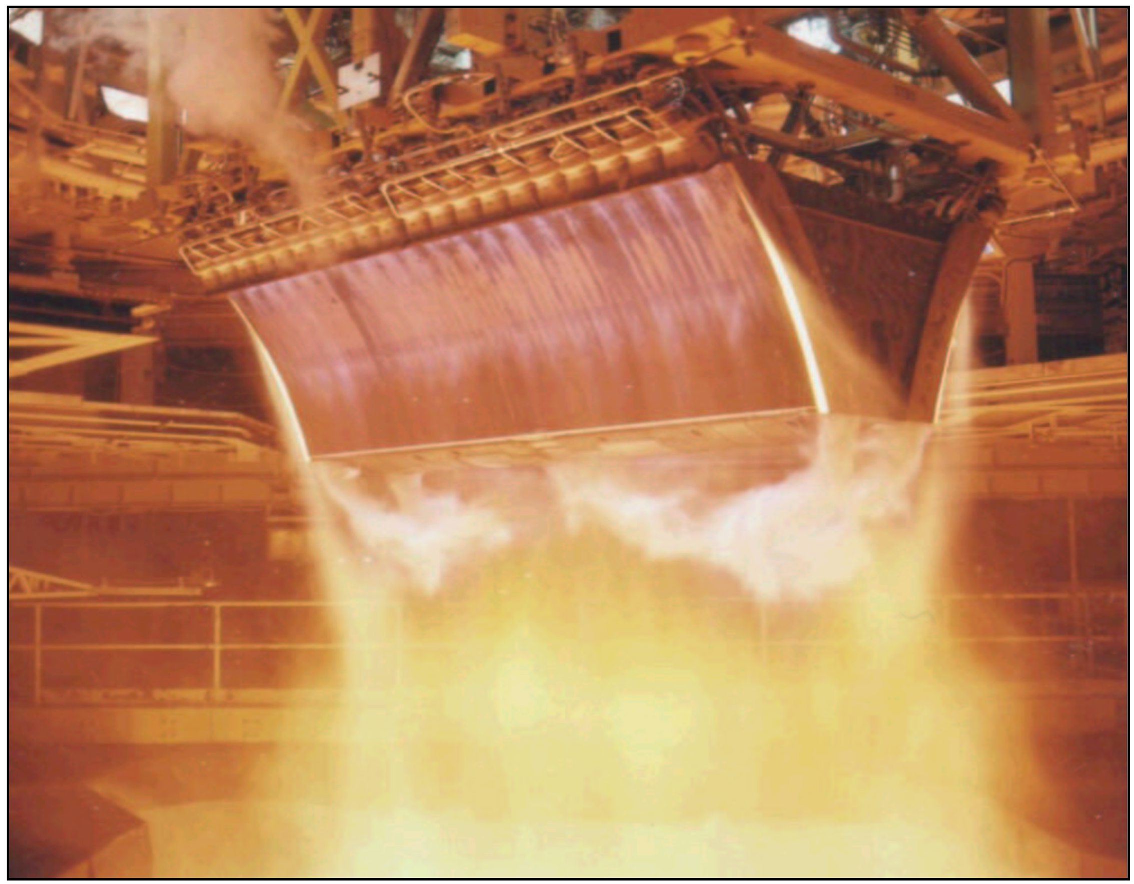
3.7% better performance
~52% of the theoretical telecoping Nozzle I_{sp} gains

SS/AA 4000

Full Scale Test of RS-2200 Rocket Engine

- **July 12, 2001**
NASA Stennis Space Center
Louisana

- **Still a Viable Option on the way to 500 sec Isp**



SS/AA 4000

Could Venture Star Actually Have Achieved SSTO?

- Compute Earth Rotational velocity at 35° (Edwards AFB) latitude

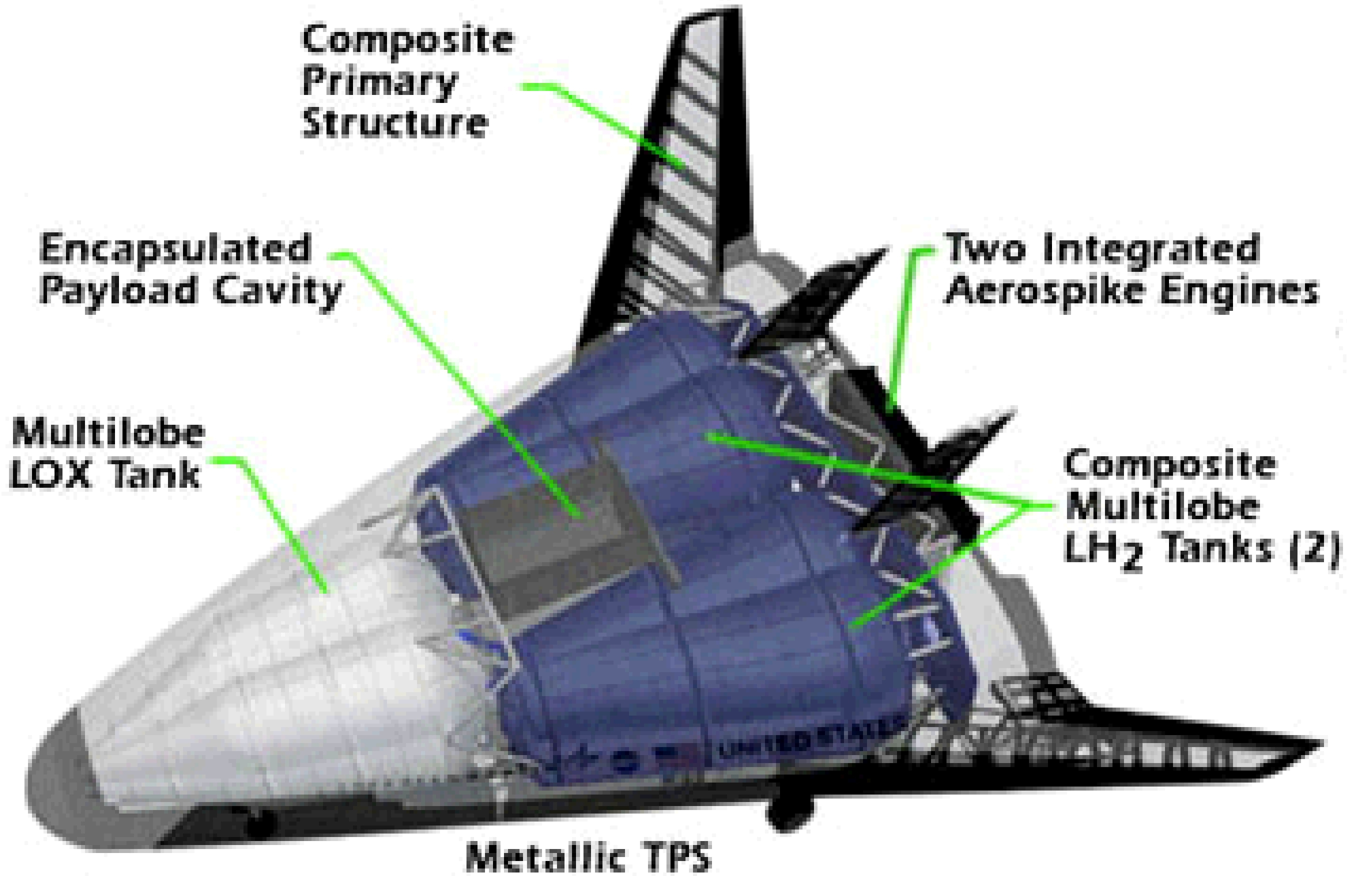
$$V_{rot_{Earth}} = \omega_{Earth} \times r_{Earth} \times \cos\left[\,\mathrm{Lat}\,\right] =$$

$$\left[0.000072722\ \mathrm{radians}\right] \times \left[6371\ \mathrm{km} \times 1000\,\frac{\mathrm{m}}{\mathrm{km}}\right] \times \cos\left[\frac{35\,\pi}{180}\ \mathrm{radians}\right] = 379.5\,\frac{\mathrm{m}}{\mathrm{sec}}$$

$$\boxed{\Delta V_{\substack{\mathrm{required}\\ \mathrm{for\ LEO}}} = 7812.3\,\frac{\mathrm{m}}{\mathrm{sec}} - 379.5\,\frac{\mathrm{m}}{\mathrm{sec}} = 7432.8\,\frac{\mathrm{m}}{\mathrm{sec}}}$$

SS/AA 4000

Venture-Star Fuel Capacities

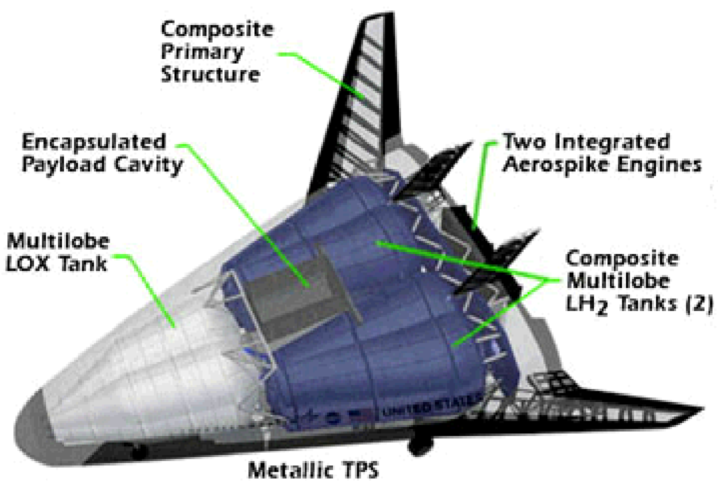


SS/AA 4000

Venture-Star Fuel Capacities

$$\left.\begin{array}{c}\text{LOX Tank Capacity: 635,000 liters}\\[2em]\text{LH}_2\text{ Tank Capacity: }2\times 900{,}000\text{ liters}\end{array}\right] \Rightarrow \boxed{\begin{array}{c}\text{TOTAL CAPACITY:}\\[1em]\text{2,435,000 LITERS}\end{array}}$$

$$\left.\begin{array}{l}\text{Max LOX Mass: }635{,}000\text{ liters}\times 1.14\,\dfrac{\text{kg}}{\text{liter}} = 723{,}900\text{ kg}\\[1.5em]\text{Max LH}_2\text{ Mass: }2\times 900{,}000\text{ liters}\times 0.07\,\dfrac{\text{kg}}{\text{liter}} = 126{,}000\text{ kg}\end{array}\right] \Rightarrow \boxed{\begin{array}{c}\text{TOTAL CAPACITY:}\\[1em]\text{849,900 kg}\end{array}}$$



Venture-Star Empty Weight

- **Original Specs were set at 100,000 kg**

  ... but by 2000 that had grown to ~135,000 kg

- **Target payload to LEO 25,000 kg, "dry weight"**

  .... Original Specs ---- 125,000 kg

  .... 2000 --- 125,000 kg

$GTO_{WT} = 974{,}900$ kg

only 3.6%

$GTO_{WT} = 1{,}009{,}900$ kg

SS/AA 4000

Venture Star: Propellant Mass Fraction:

- Based on original Dry mass, 100,000 kg

Circa: 1995

Propellant Mass Fraction:

$$\frac{849{,}900 \text{ kg}}{125{,}000 \text{ kg}} = 6.799$$

- Based on revised Dry mass, 135,000 kg

Circa: 2000

Propellant Mass Fraction:

$$\frac{849{,}900 \text{ kg}}{160{,}000 \text{ kg}} = 5.312$$

SS/AA 4000

Venture Star: Max ΔV Achievable:

Circa: 1995

$$\Delta V_{max} = g_0 \, I_{sp} \, ln\left[1 + P_{mf} \right] =$$

$$9.81 \times 453.3 \, ln\left[1{+}6.799 \right] = 9133.9 \, \frac{m}{sec}$$

Required ΔV: 7432.8 m/sec

Circa: 2000

$$\Delta V_{max} = g_0 \, I_{sp} \, ln\left[1 + P_{mf} \right] =$$

$$9.81 \times 453.3 \, ln\left[1{+}5.312 \right] = 8183.3 \, \frac{m}{sec}$$

SS/AA 4000

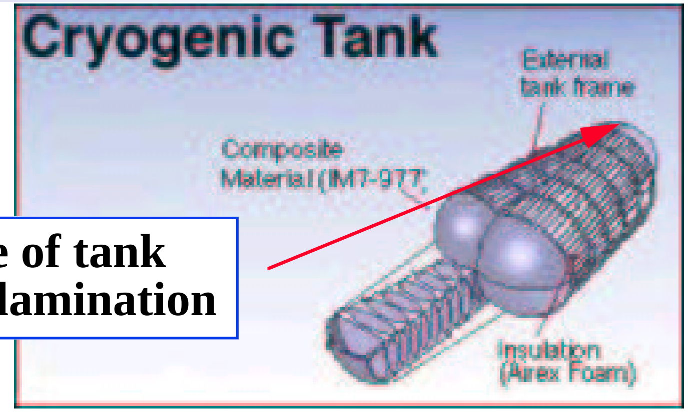
Venture Star/ X-33 : Postscript

When aerodynamic drag and engine inefficiencies are factored in ... its very unlikely that the 2000 version of the Venture Star could have achieved SSTO..... at least not with any significant payload weight .

Additional weight growth was a killer! ... that's why the composite tank rupture problems finally brought the program to its knees

SS|AA 4000

X-33: What Went Wrong?



- **LH_2 Fuel Tanks**

  Graphite/epoxy composite design intended to reduce structural weight, and withstand load of fuel and forces exerted by other X-33 structures.

- **Tank failed *after* qualification testing**

  While tank was filled with LH2 during testing air in composite structure was liquified

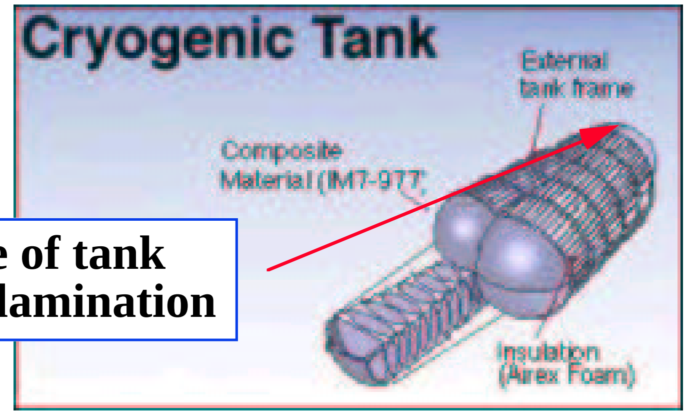
  Resulting vacuum in tank honeycomb cells caused external GN_2 purge gas to be drawn in from outside, and some gaseous H_2 was drawn in from inside

  After testing, when tank was purged of cryogenics, structured heated up, entrapped liquified air returned to gaseous state, and large pressures within the internal cells of the structure were created

  Unanticipated large internal pressures caused catastrophic de-lamination of the tank along the front lobe seam

SS/AA 4000

Venture Star: Postscript



Site of tank de-lamination

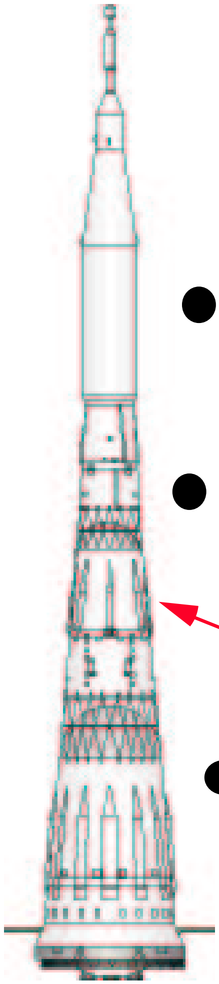
- **So for Now ... it apears the human race will have to settle for a TSTO (Two-stage-to-Orbit) RLV at best**

SS|AA 4000

Kistler K-1 RLV

SS/AA 4000

K-1 Specifications and Performance



• Kistler K-1 is a two-stage vehicle projected for full reusability at both stages.

- First stage engines: Three Aerojet/AJ26-58/-59 (NK-33) LOX/kerosene engines with a total sea level thrust of 1,020,000 lbf.
- Second stage engines: One Aerojet/AJ26-60 (NK-43) LOX/kerosene engine with a total vacuum thrust of 395,000 lbf.

NK-33/34 engiines developed for Soviet Manned Lunar Program

- K-1 vehicle gross liftoff weight of 841,000 lbm (382,300) kg

First stage: 551,000 lbm (250,500) kg
Second stage: 290,000 lbm (131,800) kg

Soviet N1F Sr

SS|AA 4000

NK-33 Engine Specs

Sea level Thrust (lbf)
339900.0

Vacuum Thrust (lbf)
378300.0

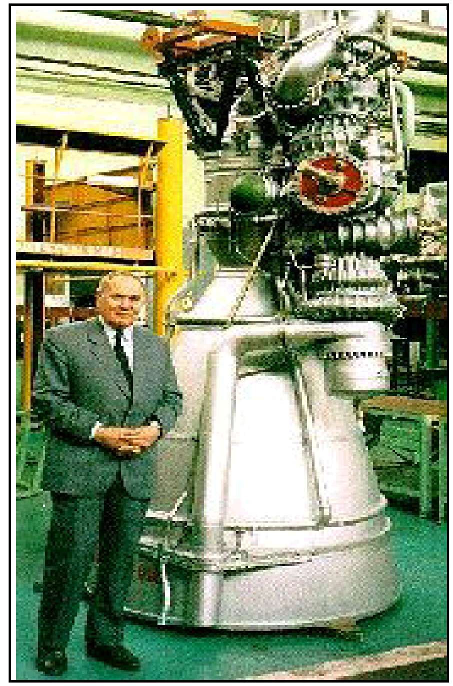
Vacuum Isp
331.30

Aexit (ft^2)
19.00457

A* (ft^2)
0.70382

Ae/A*
0.03703

A*/Ae
27.00189



Mdot (lbm/sec)
1141.865378

Pexit (psf)
800.00

Po (psia)
2109.000

To (deg F)
4144.350

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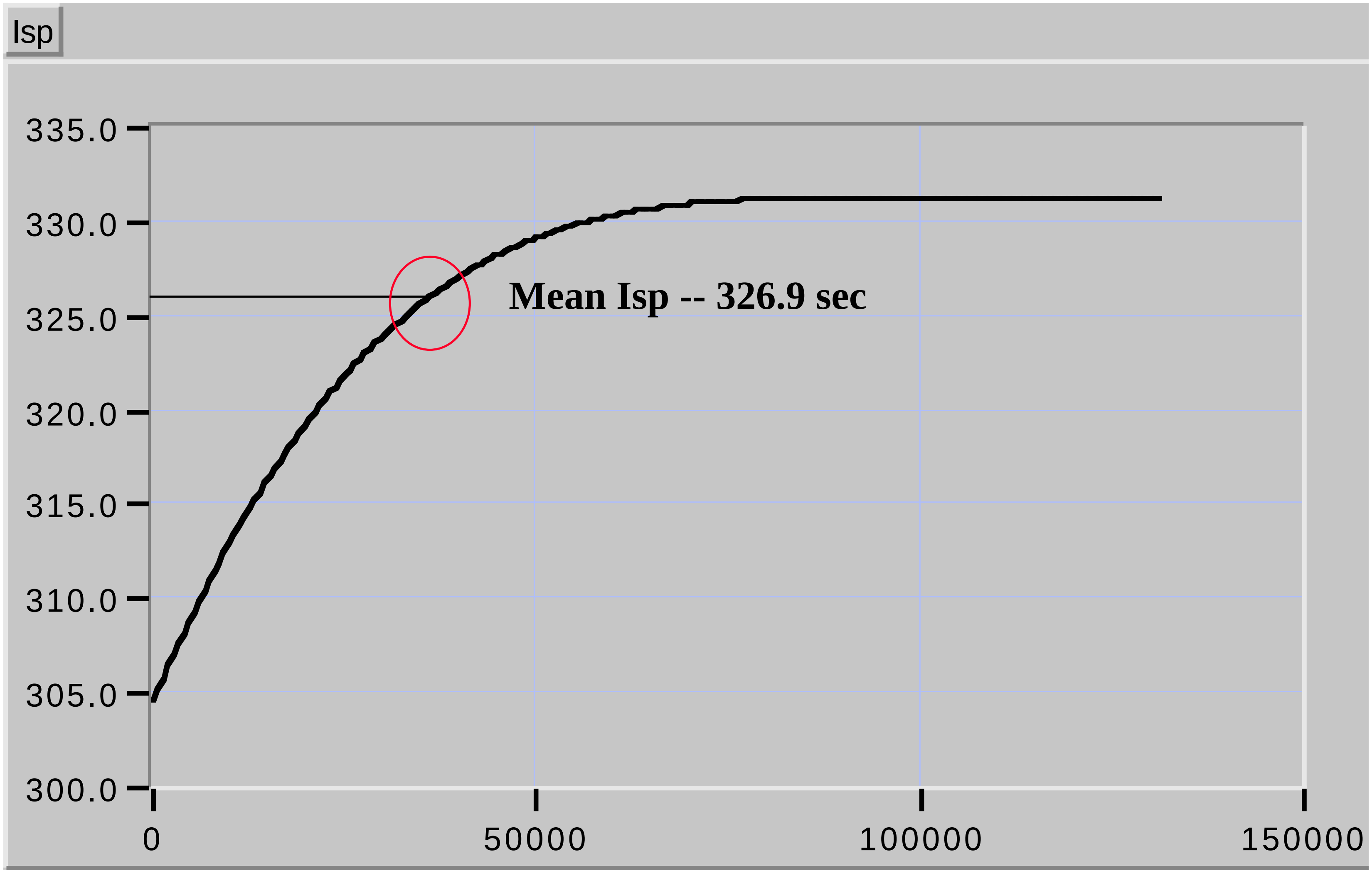
NK-33 Engine Performance



I_{sp}, sec

Altitude

SS|AA 4000

NK-43 Engine Specs

(Designed for Vacuum Operation)

Sea level Thrust (lbf)
282757.8

Vacuum Thrust (lbf)
397700.0

Vacuum Isp
348.30

Aexit (ft^2)
52.93035

A* (ft^2)
0.66123

Ae/A*
0.01249

A*/Ae
80.04834

Version of NK-33 with higher expansion ratio nozzle for operation at altitude.

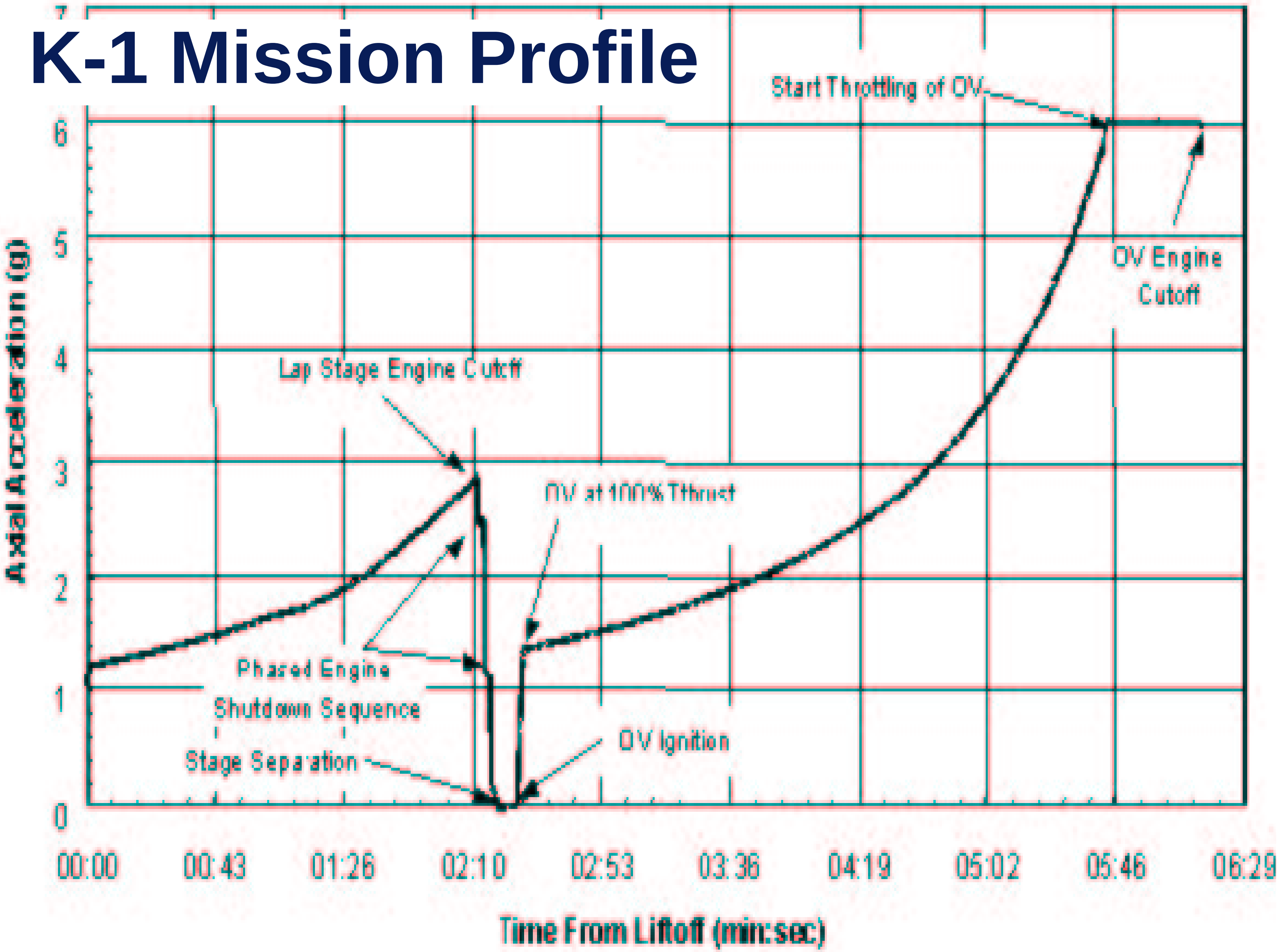
Mdot (lbm/sec)
1141.831754

Pexit (psf)
188.00

Po (psia)
2109.000

To (deg F)
4300.000

K-1 Mission Profile



Kistler K1: Stage 1 ΔV Capability

(2750 lbm payload to 900 km orbit)

$$\text{Stage 1: } P_{mf} = \frac{m_p}{M_0 - m_p} = \frac{\dot{m}_p \times T_{burn}}{M_0 - \dot{m}_p \times T_{burn}} \approx$$

$$\frac{3_{engines} \times 1140 \frac{lbm}{sec} \times 130 \text{ sec}}{2760 \text{ lbm} + 290{,}000 \text{ lbm} + 551{,}000 \text{ lbm} - 3_{engines} \times 1140 \frac{lbm}{sec} \times 130 \text{ sec}} \approx 1.114$$

$$\Delta V_{max} \approx g_0 \overline{I_{sp}} \ln\left[1 + P_{mf}\right] = 9.81 \times 326.9 \times \ln\left[2.114\right] = 2400.7 \frac{m}{sec}$$

SS/AA 4000

Kistler K1: Stage 2 ΔV Capability

(2750 lbm payload to 900 km orbit)

$$\text{Stage 2: } P_{mf} = \frac{m_p}{M_0 - m_p} = \frac{\dot{m}_p \times T_{burn}}{M_0 - \dot{m}_p \times T_{burn}} \approx$$

$$\frac{1140\ \frac{\text{lbm}}{\text{sec}} \times 210\ \text{sec}}{2760\ \text{lbm} + 290{,}000\ \text{lbm} - 1140\ \frac{\text{lbm}}{\text{sec}} \times 210\ \text{sec}} \approx 4.49$$

$$\Delta V_{max} \approx g_0\, \overline{I_{sp}}\, \ln\left[1 + P_{mf}\right] = 9.81 \times 348.3 \times \ln\left[5.49\right] = 5816\ \frac{\text{m}}{\text{sec}}$$

SS/AA 4000

Add in Earth Rotational Velocity

45 deg. inclination launch from Woomera

$$V_{rot_{Earth}} = \omega_{Earth} \times r_{Earth} \times \cos\left[Lat\right] =$$

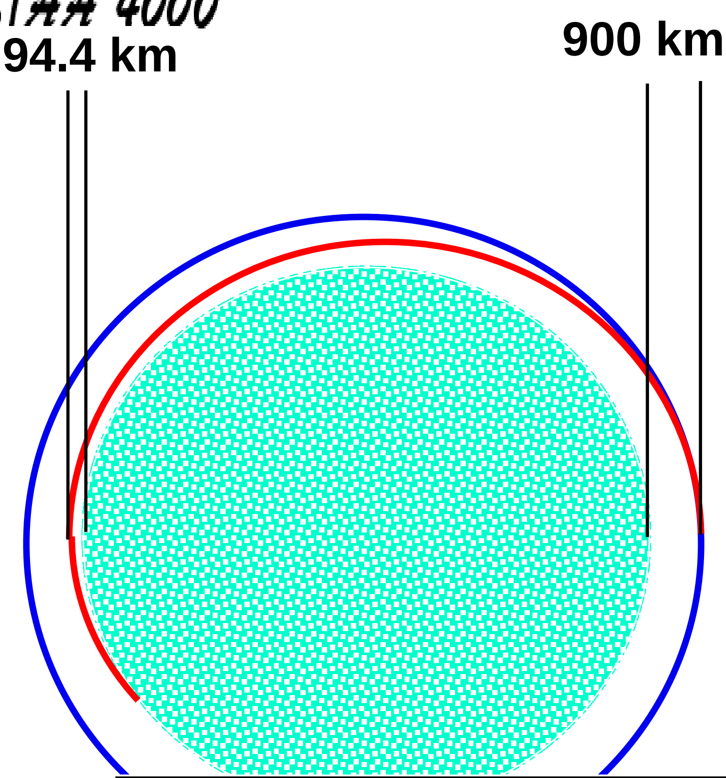
$$\left[0.000072722 \text{ radians}\right] \times \left[6371 \text{ km} \times 1000 \frac{\text{m}}{\text{km}}\right] \times \cos\left[\frac{45\,\pi}{180} \text{ radians}\right] = 3.276 \frac{\text{km}}{\text{sec}}$$

$$V_{tot} = V_{rot_{Earth}} + \left[\Delta V_{max}\right]_{stage\ 1} + \left[\Delta V_{max}\right]_{stage\ 2} =$$

$$\left[327.6 + 2400.7 + 5816\right] \frac{\text{m}}{\text{sec}} = 8544.3 \frac{\text{m}}{\text{sec}}$$

SSIAA 4000

Kistler K1: Mission Requirements



- **Stage 2 Burnout Altitude: 94.4 km**
- **Maximum Payload Altitude: (desired) 900 km**
- **Compute Transfer Orbit Eccentricity and Semi-major Axis:**

Payload to 900 km: 2750 lbm

$$e_T = \frac{r_{apogee} - r_{perigee}}{r_{apogee} + r_{perigee}} = \frac{(900 - 94.4)\,km}{(900 + 6371 + 94.4 + 6371)\,km} = 0.05865$$

$$a_T = \frac{r_{apogee} + r_{perigee}}{2} = \frac{(900 + 6371 + 94.4 + 6371)\,km}{2} = 6868.2\ km$$

Kistler K1: Mission Requirements
(concluded)

- **Compute Required Velocity at (Perigee) Stage 2 Burnout:**

$$V_{perigee} = \sqrt{\frac{2\,\mu}{R_{perigee}} - \frac{\mu}{a_T}} =$$

$$\sqrt{3.986 \times 10^5 \, \frac{km^3}{sec^2} \times \left[\frac{2}{[6371 + 94.4]\,km} - \frac{1}{6868.2\,km} \right]} = 8.079 \, \frac{km}{sec}$$

- **Max V capability of K1 to LEO ------------>** $8544.3 \, \frac{m}{sec}$

Pretty close shave (we haven't factored in drag in lower atmosphere) .... But, if they carefully optimize the trajectory... they have a reasonable chance of achieving the mission (maybe buy stock options? :-)

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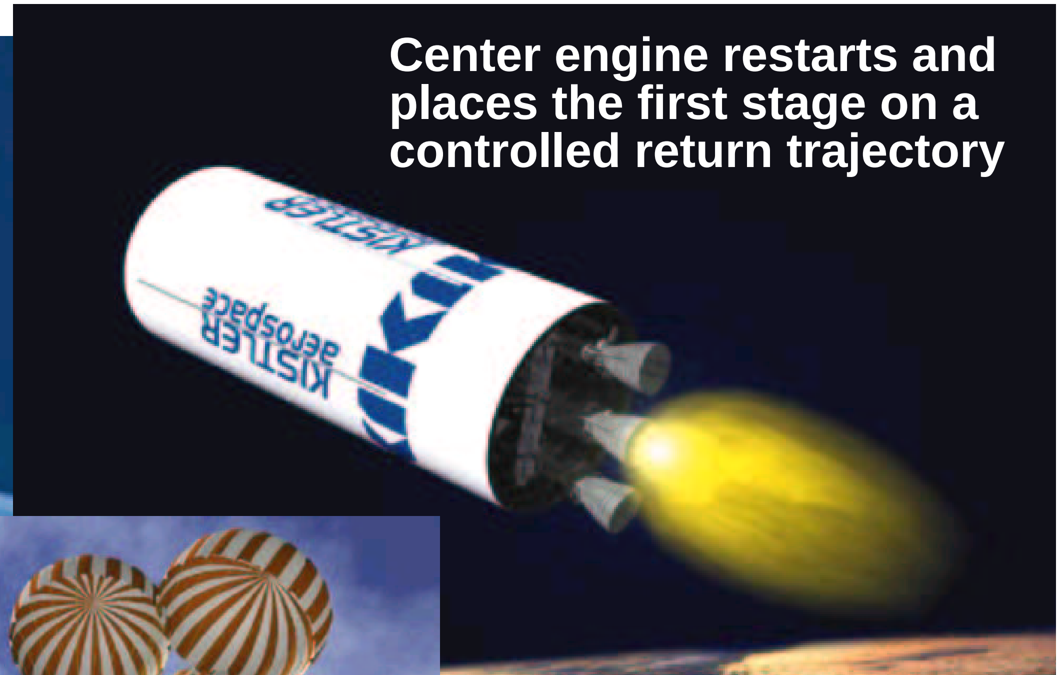
The Kistler K-1: Any Improvements Out there?



25% of Stage 1 propellant reserved for return

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What if we didn't have to reserve the 25% fuel

25% reserve

$$P_{mf} = 1.114$$

$$\Delta V_{max} \approx g_0\, \overline{I_{sp}}\, \ln[\,1 + P_{mf}] =$$
$$9.81 \times 326.9 \times \ln[\,2.114] = 2400.7\ \frac{m}{sec}$$

no reserve

$$P_{mf} = 2.363$$

$$\Delta V_{max} \approx g_0\, \overline{I_{sp}}\, \ln[\,1 + P_{mf}] =$$
$$9.81 \times 326.9 \times \ln[\,3.363] = 3900.7\ \frac{m}{sec}$$

1500 m/sec ΔV savings!

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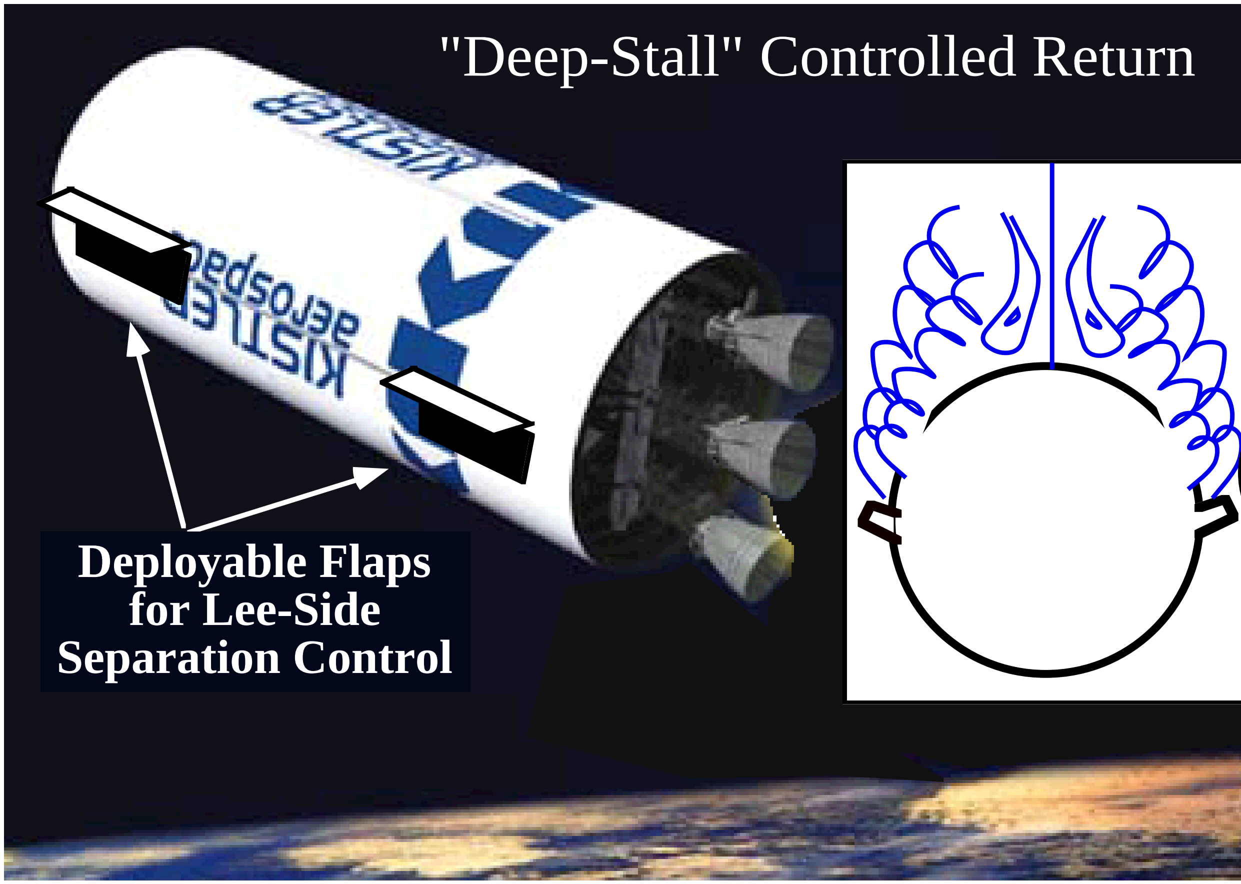
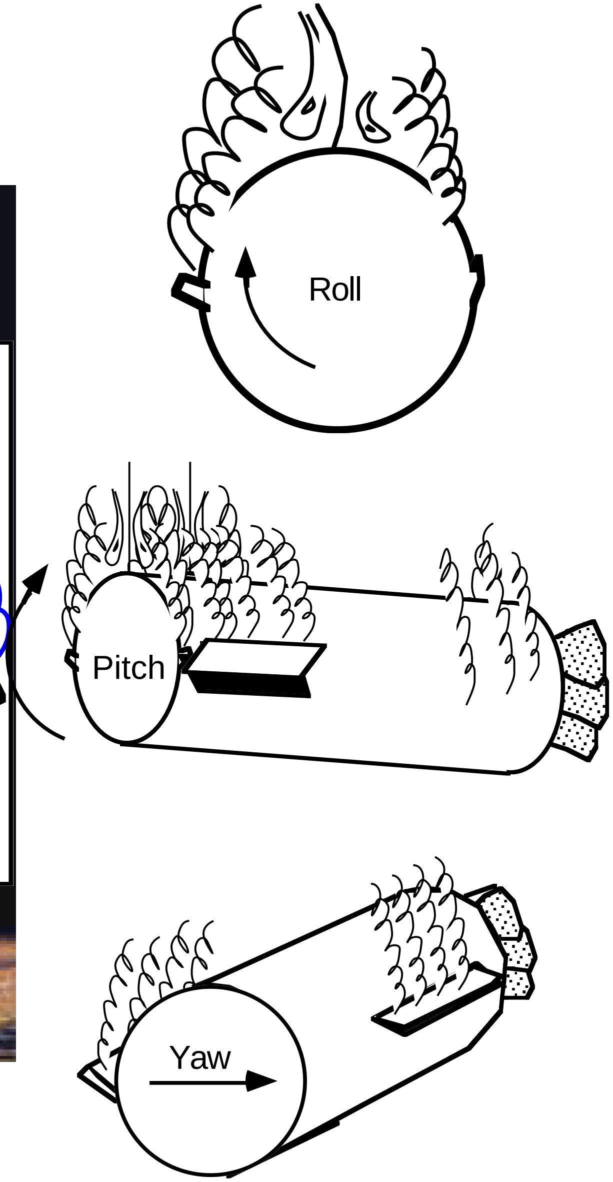
Deep-Stall Controlled Return





- **So if You want a thesis project? This is a SERIOUS Controls project!**